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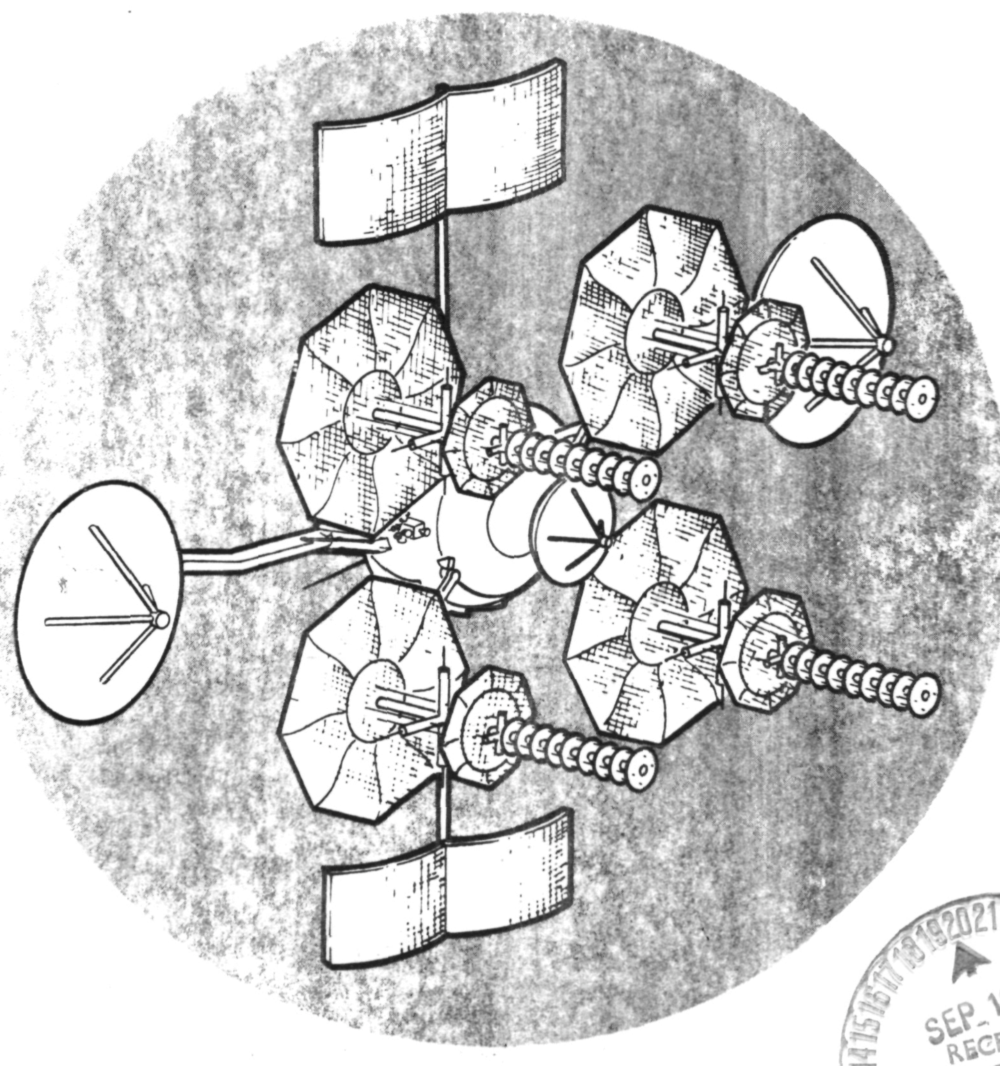
PART I FINAL REPORT

TRACKING & DATA RELAY SATELLITE SYSTEM CONFIGURATION & TRADEOFF STUDY

VOLUME I SUMMARY VOLUME



OCTOBER 1972

SUBMITTED TO
GODDARD SPACE FLIGHT CENTER
NATIONAL AERONAUTICS & SPACE ADMINISTRATION

Space Division
North American Rockwell

12214 Lakewood Boulevard Downey, California 90241

IN ACCORDANCE WITH
CONTRACT NAS5-21705

SD 72-SA-0133-1

PART I FINAL REPORT

TRACKING & DATA RELAY SATELLITE SYSTEM CONFIGURATION & TRADEOFF STUDY

VOLUME I
SUMMARY

Tom Hill

T. E. Hill
TDRS STUDY MANAGER

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FOREWORD

This report summarizes the results of Part I of the study conducted under Contract NAS5-2107. Tracking and Data Relay Satellite Configuration and Systems Trade-off Study - 3-Axis Stabilized Configuration. The study was conducted by the Space Division of North American Rockwell Corporation for the Goddard Space Flight Center of the National Aeronautics and Space Administration.

To ensure that the NASA would receive the most comprehensive and creative treatment of the problems associated with the definition of an optimum TDRS system concept, North American Rockwell entered into subcontracting agreements with the AIL Division of Cutler-Hammer and the Advanced Systems Analysis office of Magnavox. In this teaming relationship NR performed as the prime contractor with responsibility for study management, overall system engineering, TDR spacecraft and subsystem design, network operations and control, reliability engineering, and cost estimating. AIL was responsible for RF link analysis, the on-board telecommunications subsystem design and ground station RF equipment design. Magnavox was responsible for telecommunications system analysis, user spacecraft terminal design, and ground station signal processing.

The study is in two parts. Part I of the study considered all elements of the TDRS system but emphasized the design of a 3-axis stabilized satellite and a telecommunications system optimized for support of low and medium data rate user spacecraft constrained to be launched on a Delta 2914. Part II will emphasize upgrading the spacecraft design to provide telecommunications support to low and high, or low, medium and high data rate users, considering launches with the Atlas/Centaur and the Space Shuttle.

The report consists of the following seven volumes.

1.	Summary	SD 72-SA-0133-1
2.	System Engineering	SD 72-SA-0133-2
3.	Telecommunications Service System	SD 72-SA-0133-3
4.	Spacecraft and Subsystem Design	SD 72-SA-0133-4
5.	User Impact and Ground Station Design	SD 72-SA-0133-5
6.	Cost Estimates	SD 72-SA-0133-6
7.	Telecommunications System Summary	SD 72-SA-0133-7

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1.0 SUMMARY

During Part I, study efforts were directed at defining all TDRS system elements with emphasis on synthesis of a space segment design optimized to support low and medium data rate user spacecraft and launched with a Delta 2914. In addition to developing a preliminary design of the satellite, conceptual designs of the user spacecraft terminal and TDRS ground station were defined.

As a result of the analyses and design effort it was determined that a 3-axis-stabilized tracking and data relay satellite launched on a Delta 2914 provides telecommunications services considerably in excess of that required by the study statement of work. Further, the design concept, described in detail in this report, supports the needs of the Space Shuttle and has sufficient growth potential and flexibility to provide telecommunications services to high data rate users.

The spacecraft and telecommunications system uses state-of-the-art technology and equipment in all instances. This, combined with a great deal of attention to reliability, produced a very low-risk design. The major operational attributes of the system are that it:

1. Provides multiple access to both low and medium data rate users.
2. Provides two frequency options (S- and Ku-band) to medium data rate users.
3. Provides the basis for eventual support to high data rate users through the incorporation of Ku-band.
4. Provides the optimum approach to combating interference in the low data rate (VHF and UHF frequencies) channels.
5. Meets the shuttle support requirements.
6. In the event of spacecraft subsystem or telecommunications equipment failures, satellite performance degrades gracefully.

From a spacecraft design standpoint some of the major attributes, in addition to very high reliability, are:

1. A weight contingency of 82 lb (37.2 kg).
2. A power contingency at end of life of approximately 40 watts during normal telecommunications operations and a power contingency of 6 watts during emergency UHF voice service to the Space Shuttle.
3. A logical configuration for considering growth to higher power levels and larger antennas.

The telecommunications service modes provided by the 3-axis stabilized satellite design are as follows:

Modes of Service

Links	Modes of Services	Remarks
1. LDR · Return	· AGIPA Mode	Adaptive multi-beams to simultaneously service 20 LDR users with optimum signal-to interference ratio: G/T_s = -14.4dB/°K*
	· F-FOV Mode (backup)	A broad fixed beam that views all 20 LDR users: G/T_s = -28.8dB/°K*
· Forward	· Steerable Beam Mode	Provides 2 ground controlled high gain satellite steered beams for data at EIRP of +30 dBw, or data and/or voice at EIRP of +36 dBw (25% duty cycle)
	· F-FOV Mode	One of the above steerable beams can be replaced with a broad fixed beam to illuminate all 20 users for coherent ranging. The EIRP of 24 dBw can also be used for reduced data or voice.
2. MDR #1 · Return	· Dual Frequency S-band Mode Ku-band Mode	 To support current S-band users; G/T_S + 4.7 dB°K To support future high performance Ku-band users; G/T_s = 20.4dB°K
· Forward	· Data (Specification) Mode	Transmit EIRP of +41 dBw and 45.6dBw at S-and Ku-bands, respectively, at scan limits.
	· Shuttle Mode	Transmit EIRP of +47 dBw at scan limit for data at 2 kbps and 2 voice at 19.2 kbps.
· Return & Forward	· TDRS/GS Backup Mode	Antenna and/or transceiver can provide 100% functional redundancy for TDRS/GS antenna or transceiver at S- or Ku-bands
3. MDR #2 · Return	· Same as MDR #1 but does not have TDRS/GS backup capability	MDR #1 plus MDR #2 provides simultaneous support to: two S-band users; or two Ku-band users; or one S-band and one Ku-band user
4. TDRS/GS · Return	·Primary Mode	Primary mode after TDRS is "on" station; G/T_s = 6.0dB/°K
	·MDR #1 Backup Mode	Provides functional backup to primary mode at S- or Ku-band; G/T_s = 4.7 and 20.4dB/°K at S- and Ku-bands, respectively.
	·VHF Mode	TT & C subsystem provides VHF backup "on" station; becomes prime during inflight transit phases; G/T_s = -28.8dB/°K
· Forward	·Primary Mode ·MDR #1 Backup Mode ·VHF Mode	EIRP = 44.6 dBw EIRP = 41 and 45.6 dBw at S- and Ku-bands, respectively EIRP = 3 dBw
5. Tracking/ Order Wire · Return	·TDRS Tracking Mode	Provides trilateration ranging signal for TDRS spacecraft tracking and position location function when used in conjunction with 2 remote GS and main TDRS GS; G/T_s = -15.5 dB/°K
	·S-Band Order Line Mode	Provide order wire to establish priority access to MDR transponder; G/T_s = -15.5 dB°K
· Forward	·S-band Beacon Mode	Provides acquisition and tracking source for S-band MDR users with steerable antennas; EIRP = 14.5 dBw
	·TDRS Tracking Mode	EIRP = 14.5 dBw
6. Ku-band Beacon	·Beacon Mode	Provides acquisition and tracking source for Ku-band MDR users with steerable antennas; EIRP = 13 dBw
7. Frequency Source	·Slave Mode	Coherently locks on to pilot signal from GS
	·Prime Reference Mode	Becomes prime reference for telecommunication system and GS coherently locks on to Ku-band beacon

* Assumes antenna temperature of approximately 800°K; however, T in this frequency is a variable.

1.1 SYSTEM CONCEPT

Although the present tracking and data acquisition network provides a sophisticated and comprehensive support service to the earth orbital space program, it does have practical limitations with respect to spacecraft access time and information bandwidth that impose both design and operational constraints on these spacecraft. To remove the mission constraining time-bandwidth limitations of this supporting network and minimize the cost of the service, NASA is considering the implementation of a network concept entitled "Tracking and Data Relay Satellite System" that includes the use of geosynchronous communication satellites. These communication satellites will relay information to and from earth orbital spacecraft using a single primary ground station in the United States, augmented by a reduced number of remotely located ground stations.

The programs that will require support from, and benefit by, the TDRS system include scientific satellites, earth application and observation satellites, and manned spacecraft such as Space Shuttle. The needs of these users include increased coverage, higher data transmittal rates, decreased reliance on low reliability components such as tape recorders, and real time data acquisition and command capabilities, most of which are beyond the capabilities of the existing or planned ground tracking and data acquisition network.

The user spacecraft were categorized according to their range of support requirements as low data rate users (LDRU) whose data transmittal requirements are $\leq$ 10 kbs, medium data rate users (MDRU) whose data transmittal requirements are $\leq$ 1 mbs, and high data rate users (HDRU) whose data transmittal requirements are $>$ 1 mbs. The channels in the TDRS system over which the user spacecraft transmit are defined as return links, and the channels over which the user spacecraft receive are defined as forward links. Forward link requirements for LDRUs, MDRUs, and HDRUs are $\lesssim$ 1 kbs, $\lesssim$ 1 kbs, and $>$ 1 kbs, respectively. In addition, the Shuttle requirements have been specified and for the purpose of this study have been categorized as an MDRU with unique requirements such as voice and a forward link data rate of 2 kbs. During Part I, only LDRUs and MDRUs were considered. HDRU support will be considered during Part II.

Figure 1-1 indicates the general RF interfaces between the TDRS, the user spacecraft, and the ground station along with the allocated frequency bands.

Every user satellite probably will fall into the low-data-rate category ($\leq 10^4$ bps) since this link will service unsophisticated satellites, and provide emergency backup for primary links in both manned and unmanned systems. To fulfill this role it is assumed that no directivity exists on the user's antenna. The laws of physics, the current use of the radio spectrum, and the evolutionary nature of the TDRS program provide strong motivation to operate this return link in the VHF band.

Another characteristic of the LDR user is that a particular user competes with other users through his and their multipaths, as well as with almost a whole hemisphere of ground-generated RFI at the TDRS receiver. This problem could be alleviated if a clear channel could be guaranteed each user; however, this solution does not appear to be available. The approach incorporated into the baseline system to combat these problems is a combination of spectrum spreading, data encoding, and adaptive beam forming.

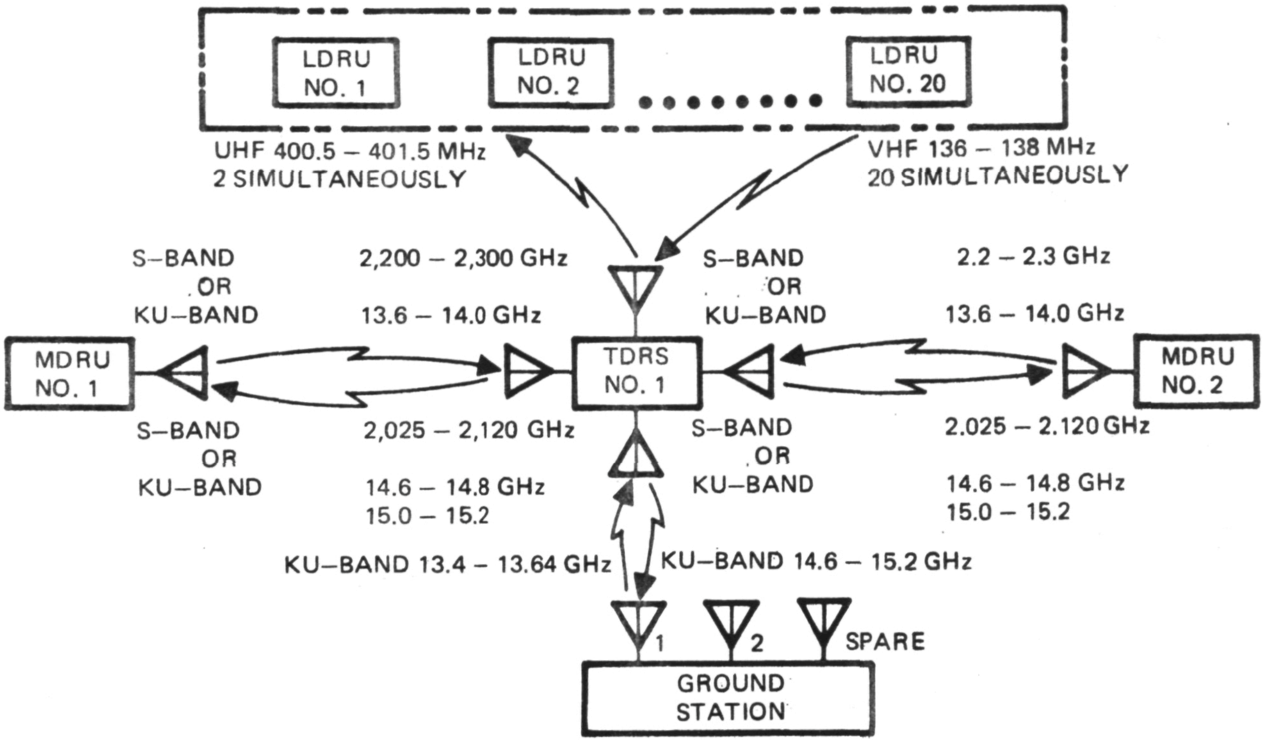


Figure 1-1. General RF Interface

The link that transfers the LDR users' data to the ground operates at Ku-band. The necessary transmitting power is small even in the face of the 17.5-db margin required in the Washington-Rosman area. At the ground, the data from all users, their multipaths, and the RFI are received in a low-noise, uncooled parametric amplifier-receiver, amplified and divided so that each element of the signal may be recovered for despreading and decoding.

In the forward link from the ground to the LDR user, a somewhat different problem exists. Data rates are low ($\leq 10^3$ bits per second), but the RFI competition at the user's receiver will vary tremendously with his location over the earth. The analysis shows the UHF band to be best when considering RFI, allocation problems, and the use of VHF for the LDR return link.

Spread-spectrum coding is used in the LDR forward link both to combat interference and to be consistent with flux density constraints.

The MDR user is characterized by higher data rates ($\leq 10^6$ bits per second on the return link), greater directivity and higher RF output. The MDR users may desire to operate at either S-band or higher frequencies. To minimize constraints on user spacecraft the TDRS design provides both forms of service. This approach also anticipates the eventual need to support high data rate users who will undoubtedly use higher frequencies such as Ku-band.

The system deployment concept synthesized during this study is illustrated in Figure 1-2, and consists of three satellites in synchronous near equatorial orbit. Two satellites are operational and are located at 11°W and 141°W longitude.

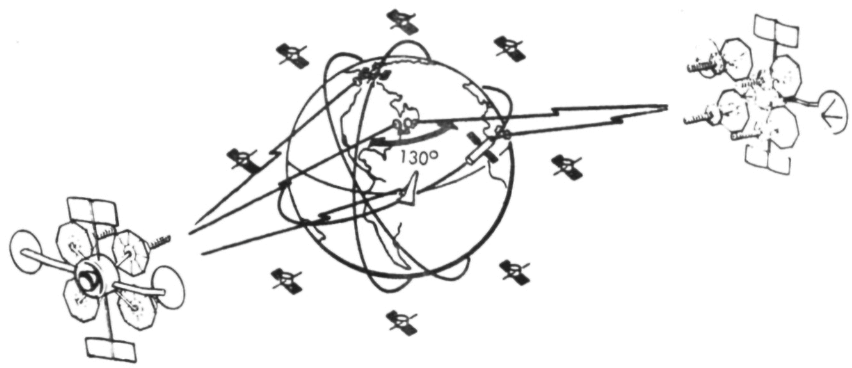


Figure 1-2. System Deployment Concept

These positions are consistent with continuous visibility to a ground station located at Rosman, N.C., with minimum elevation angles of 10°. A third satellite (not shown in Figure 1-2), is an on-orbit spare located equidistant between the two operational satellites. Each vehicle is in an orbit with an initial inclination of 2.5°(.044 rad). This approach maximizes user S/C coverage within the constraint of operating the ground station at Rosman.

The satellite designed for this deployment scheme provides telecommunications relay capability considerably greater than required by the statement-of-work specification. Each TDR satellite can simultaneously maintain:

- 2 LDRU forward links each with voice and data (spec requirement is 1 LDR forward link with voice and data)
- 20 LDRU return links
- 2 MDRU forward links at S and/or Ku-band (spec requirement is 1 MDR forward link at S or Ku-band)
- 2 MDRU return links at S and/or Ku-band (spec requirement is 1 MDR return at S or Ku-band)

Figure 1-3 illustrates the TDR satellite in its stowed and deployed configurations. The primary telecommunications relay interfaces, as indicated, are two mechanically steerable 6.5-ft (2 m) parabolic reflector antennas each of which can transmit and receive at S- or Ku-band to support the MDRUs, a four-element array of backfire antennas that transmit at UHF and receive at VHF to support the LDRUs, and a 3-ft (0.9 m) Ku-band parabolic reflector to provide the transceiver interface with the ground station. An alternative configuration was synthesized replacing one 6.5-ft (2 m) parabolic reflector with an S-band phased array. If it is determined that multiple access (greater than two simultaneously) for MDRUs is more important than high link performance, this design can be used. The resulting alternative configuration provides one high performance MDRU interface at S- and Ku-band and one multiple access interface at S-band.

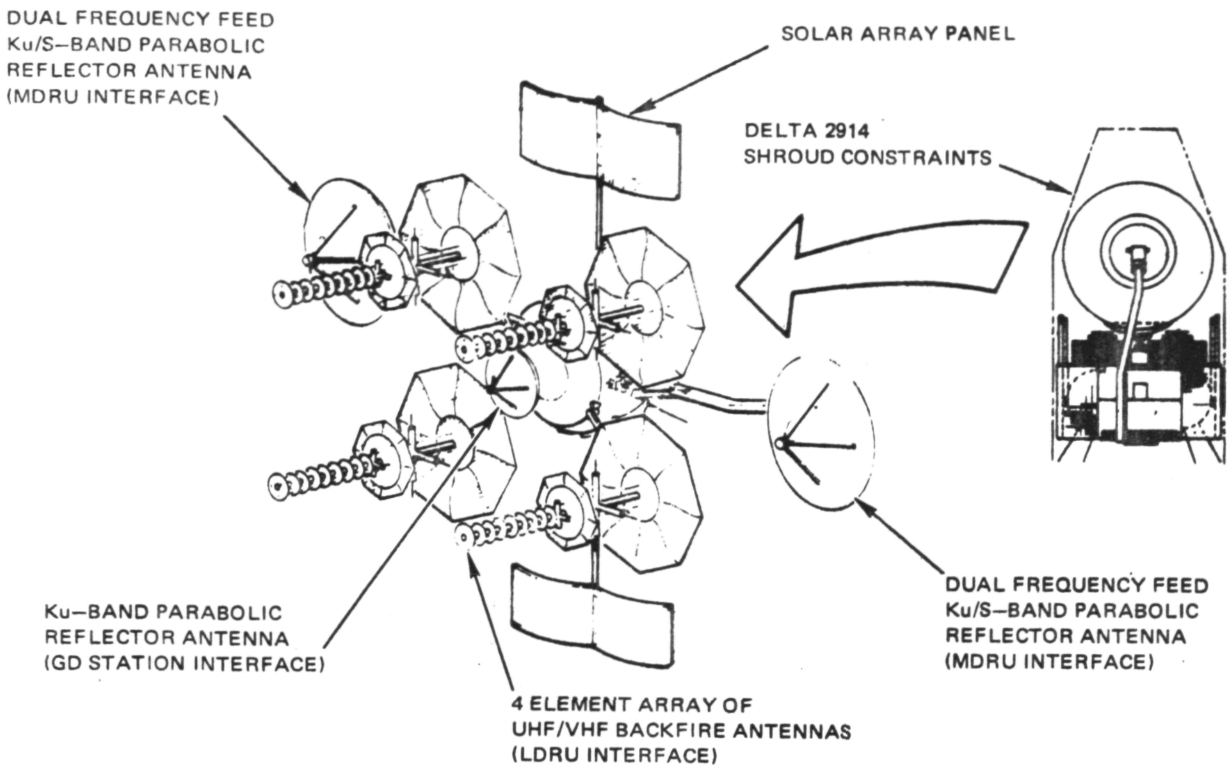


Figure 1-3. TDRS Satellite - Stowed and Deployed Configurations

In addition to the forward and return link functions previously described, the system can track all 20 LDRUs simultaneously and an S-band order wire MDR interface is provided to allow reconfiguring of the operational mode by a manned spacecraft if necessary.

The use of an orbital relay to provide the tracking and data acquisition service, while greatly enhancing the service and minimizing design and operational constraints on user spacecraft, does impose slightly different design requirements on the user spacecraft. Basically, the user terminal configuration will remain unchanged, however some additional equipment is required so that the terminals can use PN modulation and be tunable to one of four carrier frequencies. PN coding is required because of the need for:

1. Distribution of the signal energy emanating from the TDRS to the user over a bandwidth such that the signal flux density at the earth will conform to the IRAC requirements
2. Discrimination against the multipath signal which will exist in the LDR case
3. Code division multiplexing from up to 20 users on the return link per TDRS

The tunable receiver is necessary because of simultaneous visibility to two TDR satellites and compatibility with TDRS which transmits at UHF and STDN which transmits at VHF. If the STDN can be reconfigured to transmit at UHF the user S/C need only be tuneable to one of two frequencies.

A preliminary estimate of the size and prime power requirements for the user are:

		Size in^3	Size cm^3	Power (watts)
LDR	Transmitter	225	3690	16
	Receiver	195	3200	12
MDR	Transmitter	240	3930	33
	Receiver	205	3360	12

The TDRS ground station will require one 60 ft (18.3 m) antenna for each operational TDRS satellite. The receivers will use uncooled parametric amplifiers and the transmitters will use Klystron power amplifiers. Additional signal processing equipment will be required in the form of a mini-computer to analyze the LDR return link RF signals and determine the proper commands to adaptively steer the S/C VHF beam for maximum signal to interference (RFI) ratio.

1.2 SATELLITE LAUNCH AND DEPLOYMENT

The selection of the TDRS location and orbit inclination is based on three basic performance factors: payload weight, visibility of the ground station and user satellite coverage. Increased satellite spacing increases user spacecraft coverage. To increase satellite spacing, a low final TDRS orbit inclination is desirable. However, payload weight increases with increased orbit inclination, allowing for possible increases in relay capacity. Figure 1-4 shows the relationship of these basic factors. The payload capability is based on a due east launch by a Delta 2914 from KSC, with a modified Thiokol TE-M-616 apogee motor. The ground station visibility curve is based upon a minimum elevation angle at the ground station of 10° (.17 rad), and a ground station latitude of 35.2°.

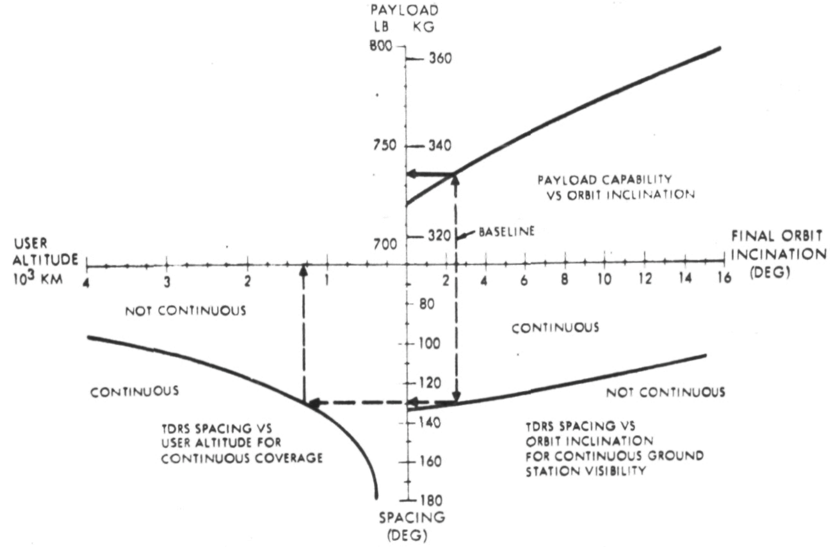


Figure 1-4. Inclination Tradeoffs

1.2.1 Deployment Analysis

As mentioned previously, the telecommunications and spacecraft designs developed in this study provide capabilities considerably greater than required

by the SOW specification for Part I. The increase in relay capabilities, provided by increasing the orbit inclination, was weighed against the increase in size of the user spacecraft cone of exclusion, and in light of the impressive relay capacity provided by the baseline design, the decision was reached to maximize user spacecraft coverage by placing the TDRS in a low-inclination orbit.

As can be seen from Figure 1-4, for a final orbit inclination of 2.5°(.044 rad), which was selected for the baseline, the TDRS can weigh 734 lb (333 kg) plus the weight of the empty apogee motor. Continuous ground station visibility can be maintained with a TDRS spacing of 130 degrees (2.27 rad). This provides continuous visibility of user satellites above 1275 kilometers (688 nautical miles). The minimum user altitude for continuous coverage increases rapidly as spacing is reduced below 130° (2.27 rad). The region where user satellites are invisible is either TDRS satellite for the baseline locations is shown in Figure 1-5 for satellite spacings of 125° (2.18 rad) and 130° (2.27 rad).

The satellite final orbit inclination is perturbed by the lunar and solar gravitational fields at a rate of approximately 0.75 degree/year. Judicious selection of launch time will cause the orbit inclination to start decreasing, pass through zero, and then increase. With an initial 2.5° (.044 rad). Therefore no north south station keeping is required.

1.2.2 Launch Analysis

The TDRS weight that can be delivered to synchronous orbit by a given booster is a function of the inclinations of the transfer orbit and of the final TDRS orbit. The apogee motor propellant and payload weights corresponding to a 2.5° (.044 rad) final inclination orbit are plotted in Figure 1-6; the maximum payload is 734 lb (333 kg) plus the 50-lb (22.7 kg) empty motor case and 8-lb (3.6 kg) burned-out insulation, and the optimum transfer orbit inclination is 27°.(.47 rad). The corresponding propellant loading for the apogee motor is 692 lb (314 kg). These values shown are the capability to synchronous orbit. The TDRS baseline operational mode injects the spacecraft into a subsynchronous drift-bias orbit and permits an increase in payload, as described later.

The TDRS is launched eastward from KSC by a Delta 2914. It leaves the transfer orbit at apogee, using a solid apogee motor. There are several tradeoffs involved in the selection of which apogee to use as the departure points. Principal factors are time and the location of each apogee relative to the final TDRS station. The TDRS is injected into the transfer orbit at the first descending node of the parking orbit. If Tananarive, Rosman, and Orroral are used to track and command the TDRS during the transfer orbits through the third apogee, visibility is maintained for the entire time except for 165 minutes at the second perigee and 38 minutes at the third perigee. If Guam is used instead of Orroral, the 165-minute period is reduced to 118 minutes. These times are considered satisfactory.

The baseline transfer orbit has an apogee at synchronous altitude and the apogee motor burns out at a velocity slightly below geosynchronous to provide an eastward drift of 5° (.087 rad)/day. The transfer orbit inclination is 27°(.47 rad). Plane changes of 1.3° (.023 rad) and 24.5° (.426 rad) are made at perigee and apogee, respectively, to provide a nominal 2.5°(.044 rad) final orbit inclination. Synchronous orbit injection occurs at the second or third apogee of the transfer orbit

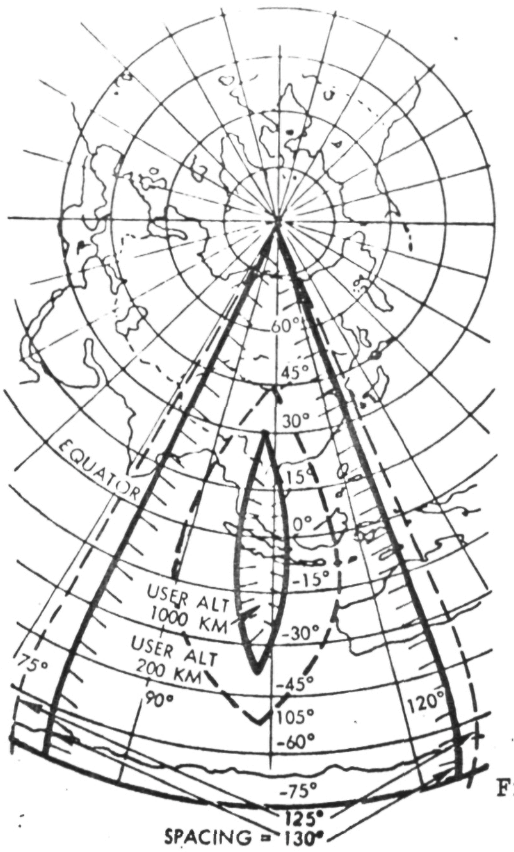


Figure 1-5. User Cone of Exclusion

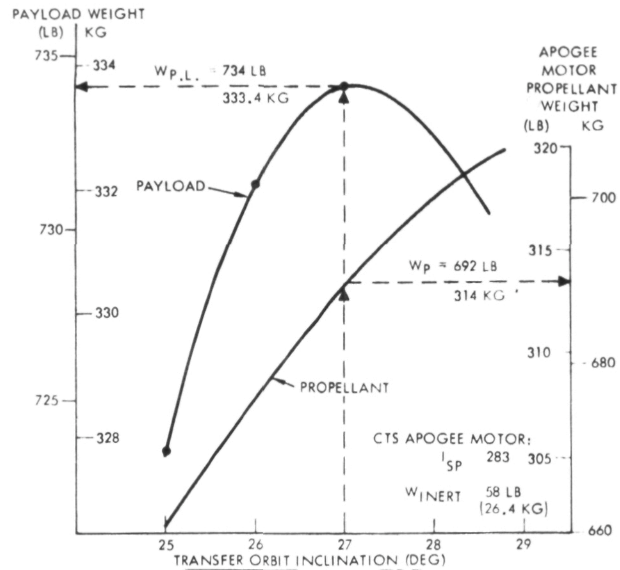


Figure 1-6. Delta 2914 Synchronous Orbit Payload

depending on the desired final location. This provides ample time for tracking, orbit determination, and vehicle precession; and places each TDRS in a position to the west of its assigned station. Figure 1-7 shows the ground trace of the TDRS during parking and transfer orbits, and the locations of the first five apogees, the tracking stations, and final TDRS locations.

The impulse provided at the transfer orbit apogee injects the TDRS into a near-geosynchronous orbit at the final orbit inclination and has both an in-plane and an out-of-plane component. The in-plane component places the vehicle in a drift orbit with an easterly motion. If the TDRS is injected at exactly synchronous velocity (drift rate equal to zero), the onboard propulsion system (monopropellant hydrazine) must initiate and stop the drift to station. If the TDRS is injected with a velocity biased slightly below synchronous, it will have a "built-in" drift, and the on-board propulsion system must only stop the drift at the appropriate station increasing the useful TDRS payload weight.

Figure 1-8 shows the effect of drift rate on (1) time to station for the three TDRS satellites, and (2) net "payload loss," apogee motor propellant reduction, and increase in on-board hydrazine. The net "payload loss" is defined as the loss in dry weight of the TDRS and is the difference between the required hydrazine and the reduced apogee propellant after the "drift bias" mode has been selected. For the drift range considered, the effect on payload is negligible. In using the drift bias mode, an actual gain in dry payload of approximately 4 pounds (1.8 kg) results.

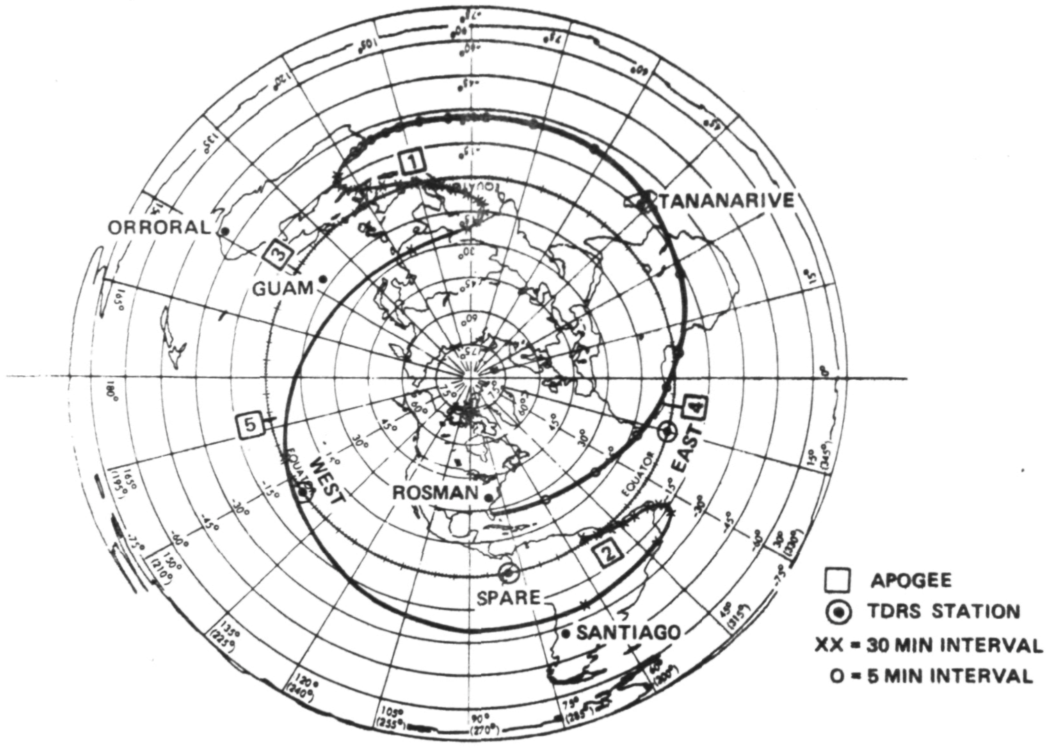


Figure 1-7. Ground Trace and Final Stations

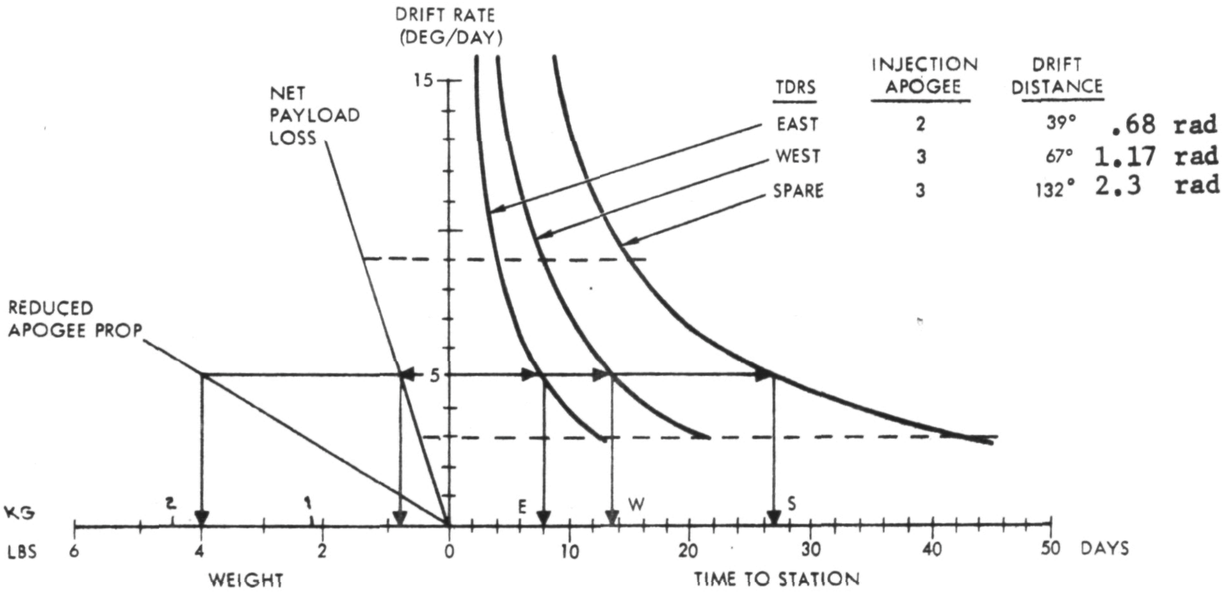


Figure 1-8. Drift Rate Effect on Payload and Drift Time

The east TDRS enters its drift orbit on the second apogee, 39° (.68 rad) west of its destination, with an eastward drift of 5° (.087 rad)/day. It arrives at its station in about 8 days. The west TDRS enters the drift orbit on the third apogee, 67° (1.17 rad) west of its destination, with the same drift rate. It arrives at its destination in 13 days. The spare enters its drift orbit on the third apogee, 132° (2.3 rad) west of its storage location, which is midway between the other two satellites. It drifts for 27 days.

An additional factor that must be considered in establishing the on-orbit payload weight is the reduction in apogee motor propellant due to on-board propellant (hydrazine) consumed during transfer orbit that need not be injected into synchronous orbit. This reduction in apogee propellant can be partially converted into payload.

The Delta 2914 injects 1490 lb (678 kg) into a 27-degree inclination transfer orbit. Six pounds (2.7 kg) of propellant are used for precession and nutation damping during transfer, leaving 1484 lb (673 kg) at synchronous orbit injection. This requires 692 lb (314 kg) of propellant and results in 1484 - 692 = 792 lb (359 kg) of payload. Included in the payload is 50 lb (22.7 kg) of empty motor case and 8 lb (3.6 kg) of burned-out insulation. The resulting 734 lb (333 kg) of useful payload is injected into synchronous orbit. However, the "drift bias" mode allows a reduction in apogee propellant of 4 lb (1.8 kg) and an increase in payload of 4 lb (1.8 kg). This results in the following propellant and payload values into the 5-degree/day drift-bias orbit.

Payload = 734 + 4 = 738 lb (334.8 kg)
Apogee propellant = 692 - 4 = 688 lb (312.2 kg)

1.2.3 Launch and Deployment Profile

Based on the TDRS deployment philosophy previously discussed, a baseline flight profile was established. Each operational satellite is injected at the apogee most convenient for eastward drift to its station. Locations of the first three apogees in such case are at 104 degrees, 306 degrees, and 148 degrees longitude. The spacecraft/launch vehicle combination injects into a 100 nmi circular inclined orbit and at the first descending node is injected into a Hohmann ellipse to synchronous altitude. At some apogee passage (second or third) of the transfer orbit the spacecraft is injected into a near-circular equatorial orbit, i.e., the thrust simultaneously removes the eccentricity and inclination of the transfer orbit, leaving slight residuals resulting from non-perfect systems performance. These residuals are removed by a vernier propulsion correction system.

Figure 1-9 illustrates the total launch profile for launch and deployment into operational status. The total mission is divided into three phases: (1) boost, (2) transfer orbit, and (3) preoperational synchronous orbit phase.

Each TDRS is launched from ETR by a Delta 2914 with a TE-364-4 third-stage at a launch azimuth of 90° (1.57 rad). The vehicle lifts into a parking orbit at a nominal altitude of 100 n mi (185 km) with an inclination of approximately 28.3° (.5 rad). The fairing is jettisoned about 36 seconds after Stage II ignition and 4 minutes before the first Stage II cutoff command and start of the parking orbit coast phase. Coast lasts 16.22 minutes and concludes when the vehicle reaches the first descending node (first perigee) at 3 degrees east longitude. At the node, the second stage restarts for 28 seconds. After its burnout, the third stage and the TDRS are spun up to 90 rpm, Stage III ignites and burns for 24 seconds to complete transfer orbit insertion. Payload separation occurs two minutes later and the TDRS remains spinning until after insertion into synchronous orbit.

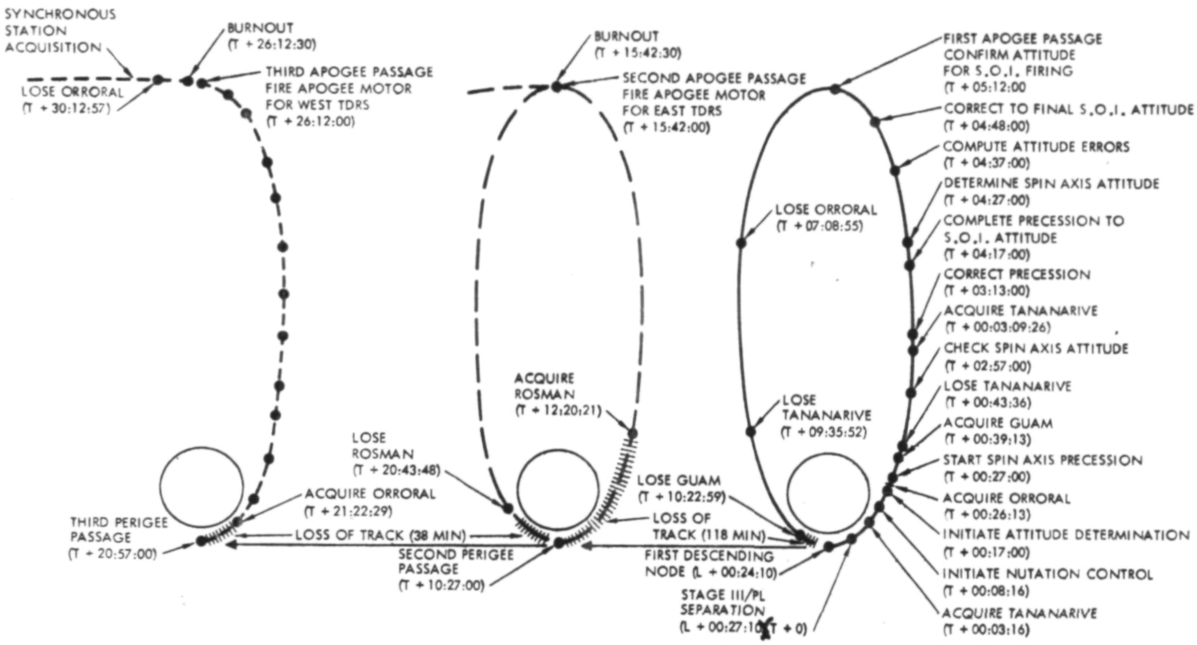


Figure 1-9. Overall Launch and Deployment Profile

After payload separation (~ 110 n mi/204 km) the spacecraft coasts to synchronous altitude in an elliptical transfer orbit. The long transit allows time for smoothing and processing of tracking data and for reorienting the spacecraft for the apogee motor burn. The transfer time from injection (perigee) to first apogee (one-half orbit) is 5.25 hours. During the entire transfer orbit, the spacecraft will be spinning and will maneuver into appropriate attitudes for attitude determination and measurement, and nutation will be damped out.

At the given apogee, the apogee motor fires to change plane and circularize the orbit at synchronous altitude for approach to operational station. Complete transfer orbit time is approximately 15.75 hours to the second apogee and approximately 26.25 hours to the third apogee, sufficient for all required operations and economic fuel consumption.

After apogee motor burnout, the spacecraft is despun and stabilized (momentum wheels spun) in an essentially equatorial orbit. The solar panels are deployed 1.5 hours after spacecraft despin and the antennas are deployed 20 minutes later. The spacecraft then acquires the sun and earth and achieves near-continuous sunlight for the mission at synchronous altitude. The spacecraft drifts to its assigned station. Appropriate post-apogee delta-V maneuvers are performed to correct the spacecraft injection errors and to acquire the proper drift orbit (about 24 hours after apogee motor burnout).

1.3 TELECOMMUNICATIONS DESIGN

The major constraints that impacted the TDRS telecommunication design are the capabilities of the Delta 2914, the radio frequency interference (RFI) and multipath environment, and inclement weather at the ground station. The booster payload capability imposes a weight, power, and volume limit on the spacecraft which correspondingly limit antenna size, system redundancy, etc. From the outset the RFI and multipath phenomena presented the single most important technical problem area. The use of pseudonoise (PN) modulation concepts and the Adaptive Ground Implemented Phased Array (AGIPA) offers a viable solution to these problems associated with supporting LDRU's.

The basic design goal was to provide a viable, cost-effective telecommunication service to a variety of user spacecraft. This requires a system that can provide a multiple semi-random access service to LDRU and MDRU spacecraft.

Telecommunication support required by the users as per the SOW is presented in Table 1-1. The table is divided into three basic categories; namely, LDR user, MDR user, and a manned user which is the Space Shuttle. In addition to those presented in the table, the ground station/TDRS link is required to operate at Ku-band, with a rain margin of +17.5 dB. Where there are multiple frequencies shown in the table, separated by the word/logic "or", the option was given to the contractor to select one or more of these frequencies to optimally support that channel's performance requirements.

Each TDRS satellite must simultaneously support 20 LDR users and at least one MDR user in the return link (user to TDRS to GS) and one LDR user and at least one MDR user in the command link (GS to TDRS to user). However, throughout this study it has been a goal, both in the design of the telecommunications subsystem and of the spacecraft to (1) provide simultaneous support capabilities greater than those required, (2) to maximize system flexibility and adaptability to user needs, and (3) to minimize risk.

1.3.1 Telecommunications System Analysis

The low data rate service uses a UHF command link with two TDRS electronically steered beams and a relatively high gain antenna. The LDR transponder provides an alternate capability for a F-FOV approach where all users are simultaneously illuminated at an EIRP of +24 dBw. This mode can be used to coherently lock all users simultaneously or to transmit command data or voice at reduced capability. System EIRP at UHF is +30 dBw/beam. The return link employs the adaptive ground implemented phased array (AGIPA) concept which provides 20 independent beams (one for each user). The user spacecraft are discriminated by use of unique PN codes in the return link. The AGIPA concept provides an adaptive spatial filtering of RFI, and an optimization of the signal-to-interference ratio. The system is designed such that in the event of failures it will degrade gracefully to a fixed field-of-view configuration. The key design features of the low data rate services are shown in Table 1-2.

The MDR user telecommunications service has a dual frequency feature; namely, an S-band link to support current MDR users and a Ku-band link to support future MDR users with higher performance. Two MDR users can obtain simultaneous support via two 6.5 ft (2 m) dishes on the TDRS. In addition, the MDR service can support manned users such as Space Shuttle. The system

Table 1-1. Telecommunications Service Requirements (as per SOW)

Description	LDR User	MDR User	Manned User (Shuttle)
Number of users	Forward: 1 (minimum) Return: 20	Minimum of one	Minimum of one (Replaces requirement for one MDR)
Frequency:	Forward: VHF or UHF or S-band Return: VHF	S- or X- or Ku-band	S-band VHF-band
Communications requirement	Forward: 100 to 1000 bps Return: 1 to 10 kbps	Forward: 100 to 1000 bps Return: 10 to 1000 kbps	Forward: 2 kbps, 1 or 2 voice at 19.2 kbps Return: 76.8 kbps, 1 or 2 voice at 19.2 kbps
Constraints	Linear transponder in return link High RFI Flux density (IRAC) VHF $\leq$ -144 dBw/m^2/4 kHz UHF $\leq$ -150 dBw/m^2/4 kHz S-band $\leq$ -154 dBw/m^2/4 kHz EIRP = +30 dBw/channel for UHF & VHF +41 dBw/channel for S-band BER = 10^{-5}	Linear TDRS transponder in return link Variable frequency Flux density (IRAC) S-band $\leq$ -154 dBw/m^2/4 kHz X-band $\leq$ -150 dBw/m^2/4 kHz Ku-band $\leq$ -152 dBw/m^2/4 kHz BER = 10^{-5}	User antenna gain = +3 dB BER Voice: 10^{-3} Data: 10^{-4} User transmit power = 16 dBw

Table 1-2. Key Design Features of the Low Data Rate Service

Forward link	Frequency band Polarization EIRP (at 31° (0.54 rad) FOV)	400.5 to 401.5 MHz Circular Steered beam 30 dBw data/36 dBw voice @ 25% duty cycle F-FOV 24 dBw
Return link	Frequency band Polarization G/T_s AGIPA G/T_s FFOV	136 to 138 MHz Linear 2 planes -14.4 dB/°K* -18.8 dB/°K*
Antenna configuration		Backfire antenna 4-element array
Transceiver configuration		Frequency translating

* Assumes nominal antenna temperature of 800°K; however, T in this frequency is a variable.

antenna tracking consists of open-loop tracking at S-band and autotract at Ku-band. Furthermore, the MDR antenna system and/or transceiver serves as backup for the GS/TDRS link at either S- or Ku-band. The MDR service is essentially a "bent pipe" on the return link providing support for a variety of MDR users regardless of the signal formats employed. The MDR service features are shown in Table 1-3.

Table 1-3. Key Design Features of the Medium Data Rate Service

Forward link	Frequency EIRP Polarization	S-/Ku-band S-band data = 41 dBw; voice = 47 dBw at 25% duty cycle Ku-band 45.6 dBw Circular
Return link	Frequency G/T_s Polarization	S-/Ku-band 3.9/20.4 dB Circular
Antenna configuration		2 S- and Ku-band Parabolic reflectors
Transceiver configuration		Frequency translating

1.3.1.1 The Interference Problem

The TDRS and low data rate user spacecraft will be confronted with four basic types of interference:

1. Unintentional, upward directed, radio frequency interference (RFI) originating from communications equipment located on earth.

2. Background interference (i.e., "trash noise") originating primarily in urban areas and is the composite of such things as ignition noise, switching transients, corona, and other forms of man-made noise.

3. Multipath interference between the direct path signal in the USER/TDRS link and a replica of that signal reflected off the earth.

4. Co-channel interference between one user and all the other users in the same band. This is peculiar to the return link where all users are simultaneously accessing TDRS through the same channel.

Unintentional, upward directed RFI from emitters located on the earth and in view of both the TDRS and user spacecraft can be a deleterious source of interference. Frequency bands considered in evaluating the impact of RFI on the TDRS system are:

Forward Link	Return Link
117.5 MHz ($\pm$25 kHz)	136 to 138 MHz
127.7 to 127.85 MHz	
148 to 150 MHz	
400.5 to 401.5 MHz	

An estimate of the worst case RFI distribution at the TDRS and user was derived. Figure 1-10 presents a graphical indication of the RFI power density at the TDRS located at 11 degrees west longitude over the frequency band 117 to 155 MHz. The band 144-148 MHz is the amateur band and has not been modeled.

The impact of RFI on the user spacecraft was assessed. Two user locations were used, 50 degrees north/30 degrees east and 38 degrees north/85 degrees west, and the RFI density levels at the user spacecraft are shown in Figures 1-11 and 1-12, respectively. In each case the user spacecraft is equipped with an omni-directional antenna and is at an altitude of 1000 kilometers. As an indication of the coverage of this user spacecraft, a satellite at 1000 km centered at 38 degrees north and 88 degrees west will, with an omni-directional antenna, be in view of all of CONUS, and parts of Mexico and Canada.

As can be seen the estimated RFI levels are extremely high throughout this portion of the spectrum. Multipath of 7 db was used in all calculations.

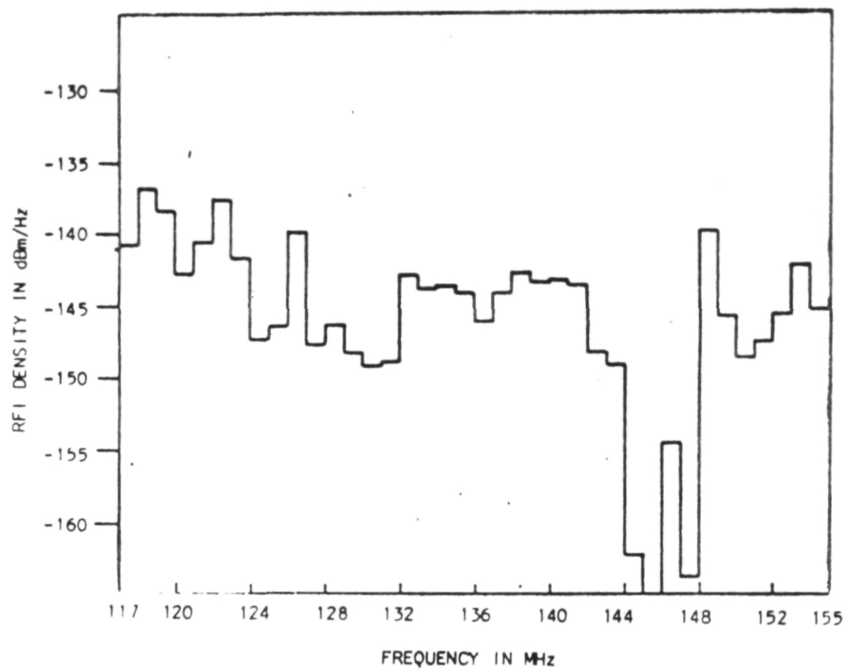


Figure 1-10. RFI Power Density for TDRS Located at 11° W

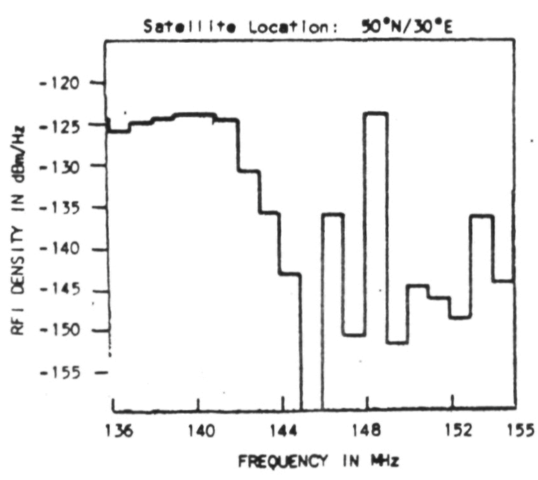


Figure 1-11. RFI Power Density at User Spacecraft (1000-km Altitude and Omnidirectional Antenna - Satellite Location 50°N/30°E)

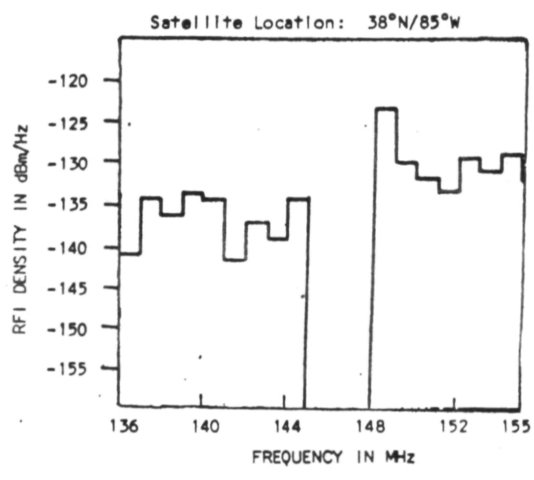


Figure 1-12. RFI Power Density at User Spacecraft (1000-km Altitude and Omnidirectional Antenna - Satellite Location 38°N/85°W)

The remaining band of interest is in the 400.5 MHz to 401.5 MHz range. An estimate was compiled, through the International Frequency List, of the number of potential RFI sources in this band and is presented in Table 1-4 for the various ITU regions. This data shows that RFI in this portion of the spectrum can be expected to be considerably less severe than in the VHF area.

Table 1-4. RFI Sources in the Band 400.5 to 401.5 MHz

ITU	Emitter Power on the Ground (watts)				
Region No.	0 - 0.9	1 - 9	10 - 99	100 - 999	≥1000
I	59	4	1	1	1
II	61	1	-	-	-
III	62	12		10	

Another important source of noise was considered and evaluated for its impact on TDRS/user channel. This interference is the sum total of the "trash"--a more or less continuous background noise which originates in an urban environment and is due primarily to such things as ignition noise, switching transients, corona, etc. This continuous low level interference will interfere with the desired signals in the command band, and can have a deleterious effect on the antenna temperature at the user spacecraft.

Figure 1-13 depicts the "trash noise" density as a function of urban radius for various frequency bands and user spacecraft altitudes. Due to the large field of view of even low orbiting satellites, the urban radius can be large. This is the result of the appearance of many discrete point sources acting as one large area distributed source. Radius values can range from 200 to 400 km for the New York area and 100 km for the Chicago area, to negligible for water or desert areas.

A signal transmitted from either TDRS or user satellite arrives at the other satellite by direct and indirect (reflection off a non-smooth earth) paths. This indirect path is called the multipath signal.

The reflected power can consist of both specular and diffuse components. The specular component is essentially a delayed replica of the transmitted signal, whereas the diffuse component is noise-like. The relative expected specular and diffuse reflected power are illustrated in Figure 1-14 as a function of the grazing angle ψ for 136 MHz and an average earth roughness and correlation distance. The specular power decreases with fequency and the diffuse power is essentially independent of frequency, and at low-grazing angles the divergence factor diminishes the multipath signal, while at high-grazing angles the primary reflected energy is diffuse. For reasonable

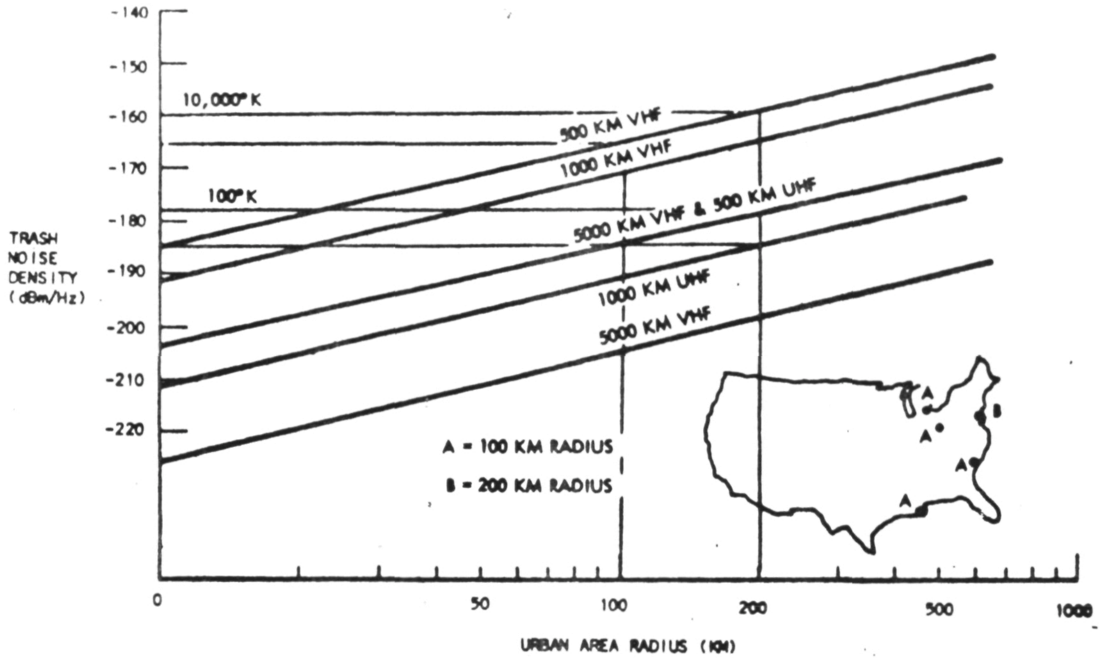


Figure 1-13. Trash Noise at User Spacecraft

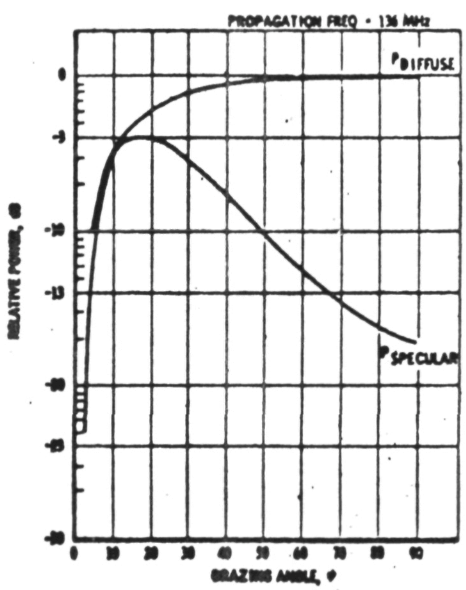


Figure 1-14. Relative Specular and Diffuse Reflected Power Versus Grazing Angle

roughness factors and correlation lengths, the primary source of reflected power is diffuse for grazing angles in excess of 20 degrees at VHF or UHF. Figure 1-15 indicates the multipath signal level as a function of orbital altitude of the user satellite.

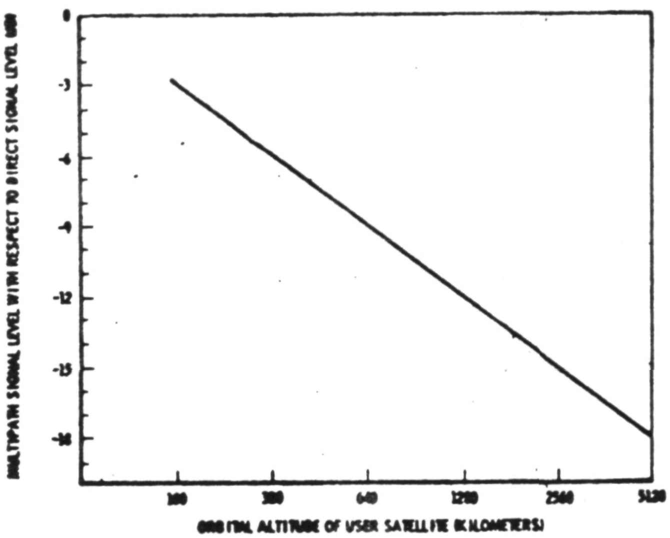


Figure 1-15. Multipath/Direct Signal Ratio as a Function of Orbital Altitude

In the forward link, the plot of Figure 1-15 can be used directly to determine the multipath impact on the user spacecraft; however, for the return VHF link, 20 user signals are accessing a TDRS simultaneously. Obviously the total multipath signal in the field of view of a TDRS will depend upon the user altitudes. Table 1-5 lists the projected satellite populations for the low altitude user spacecraft from 1976 through 1980.

Figure 1-16 shows the expected total multipath contributed by the total satellite population by year. Summing up the multipath signal contributed by 20 simultaneous users in 1976, the total multipath relative to one user is +7 dB. Thus, the multipath problem in the return link is diminished from the worst case condition where all 20 users are considered to be in low orbits; e.g., 300 km.

In addition to RFI and multipath signals, a return link signal will be subjected to interference from signals originating from the 19 other user spacecraft and propagating along the direct path from user-to-TDRS. Their effect will be much more pronounced than that of the return link multipath signals with an interference/signal ratio on the order of 12 dB.

Table 1-5. Low Altitude Spacecraft Population Projections: 1976-1980

Altitudes	Year				
	1976	1977	1978	1979	1980
150 - 276 km	1	1			
275 - 300	4		1	1	1
300 - 400	7	7	1	2	2
400 - 600	9	9	4	2	
700 - 800	10				
800 - 900	9	8	3		
1000 - 5000	10	9	2	3	3
>5000	8		6	4	4

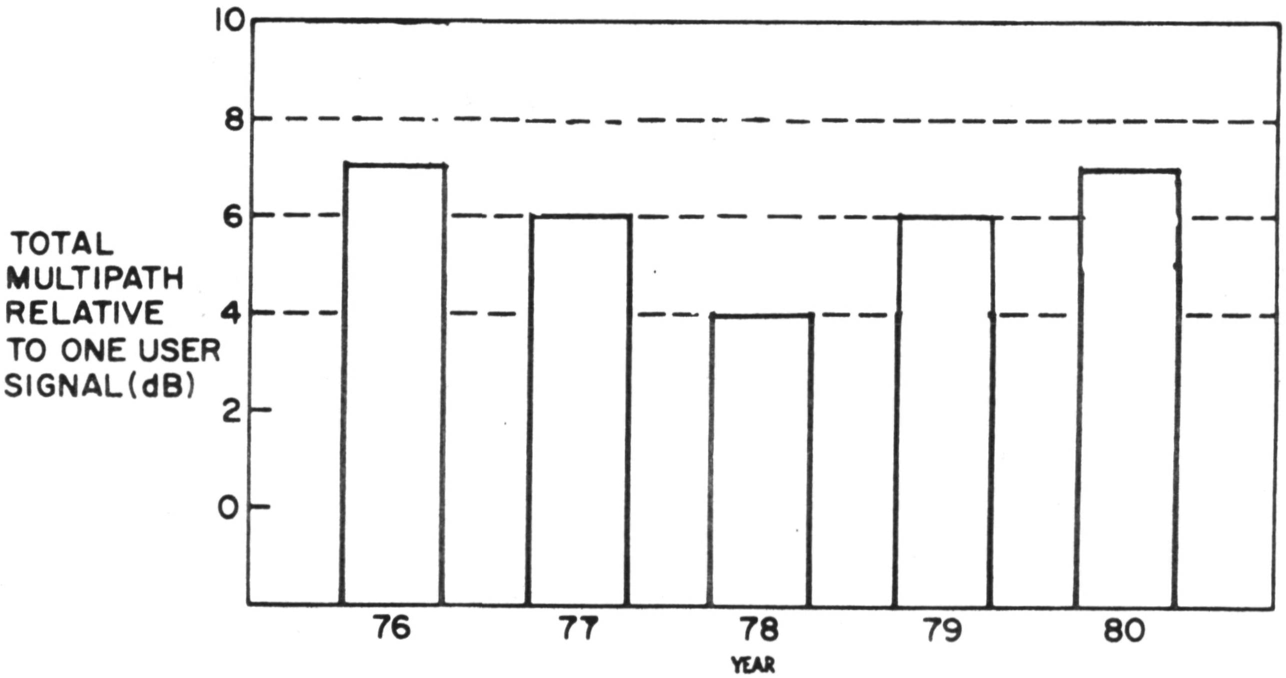


Figure 1-16. Projected Multipath Level

As implied by the foregoing discussion, the major concern in the design of the LDRU communications link was interference and the development of electromagnetic and encoding techniques to meet performance requirements in the face of these problems.

RF links to the medium data rate users, although significantly influencing the size and configuration of the TDRS, need not contend with the enormous complications of RFI or multipath present in the LDRU links. The major design problem in these links is to configure antennas that will provide multiple access.

1.3.1.2 Frequency Selection

Frequency bands for both ground/space and space/space communications links were provided by the TDRS Project Office at NASA/GSFC and are shown in Tables 1-6 and 1-7.

Table 1-6. TDRSS Ground/Space Link Frequency Band Selection

Requirement	Ground to Space Link	Space to Ground Link
Primary frequency Ku-band	13.4 to 14.0 GHz	14.6 to 15.2 GHz
Backup frequency S-band	2200 to 2290 MHz	2025 to 2110 MHz
Launch phase Stationkeeping VHF	148 to 150 MHz	136 to 138 MHz

Table 1-7. TDRSS Space/Space Link Frequency Band Selection

Link	LDR User Link	MDR User Link
Telemetry	VHF: 136 to 138 MHz	S-band: 2200 to 2300 MHz
Command	VHF: 108 to 132 MHz (2 - 100 kHz channels) UHF: 400.5 to 401.5 MHz (2 - 500 kHz channels)	S-band: 2025 to 2120 MHz

During the study three LDR forward link frequency bands were investigated:

VHF	117.50 MHz (BW = 50 kHz)
	127.75 MHz (BW = 100 kHz)
UHF	401 MHz (BW = 1 MHz)
S-band	2025 - 2120 MHz (BW = 10 MHz)

The 117.5 MHz band is considered impractical because the narrow channel bandwidth (50 kHz) is inadequate to meet range error requirements. Furthermore, it does not provide enough processing gain to offset the effects of co-channel multipath. A similar objection is presented for an assignment at 127.7 MHz. In this case the 100 kHz band appears adequate; however, if it is split into two 50 kHz bands (one per TDRS), the range accuracies specified in the work statement (namely, 15 meters rms) cannot be achieved.

There appears to be less RFI at UHF than at VHF, making it more attractive for the LDR forward link. Moreover, evaluation of the "trash noise" indicates this interference is 20 dB less at UHF than at VHF (e.g., a "trash noise" temperature of 10,000 K at VHF would be ∿100 K at UHF), reducing by an order of magnitude the "trash noise" contribution to system noise temperatures. S-band was rejected because of the large implementation impact.

On the LDR return link the 136 to 138 MHz band, while possessing considerable RFI, appears least susceptible. Within this band the 136 to 137 MHz segment is preferable.

The forward and return MDR links appear to offer no particular tradeoff. The links are relatively RFI free and selection is a function of obtaining appropriate frequency assignments such that adjacent bands introduce minimum interference, and sufficient bandwidth is available for growth to support high data rate users.

The space/ground link at Ku-band has adequate bandwidth for both uplink and downlink signals in an FDM/FM mode. One major aspect of the space/ground link is the rainfall margin. Increased TDRS transmit power, increased GS antenna gain, or pre- and post-detection spatial diversity combined at the GS are viable solutions to this problem.

The final item which impacts frequency trades was the recommendations by IRAC regarding the flux density at the earth's surface from a satellite signal. For the bands of interest the IRAC specifications and the minimum spread bandwidth required to reduce the flux density to an acceptable level are presented in Table 1-8.

From the candidate frequencies and their associated channel bandwidths, the overall frequency plan shown in Table 1-9 is recommended.

Table 1-8. Bandwidth Spreading Required to Meet IRAC Specifications

Frequency Band	IRAC Required Flux Density in 4 kHz	EIRP	Minimum Spread Bandwidth
VHF	$-144\ dBw/m^2$	30 dBw	63 kHz
UHF	$-150\ dBw/m^2$	30 dBw	250 kHz
S-band	$-154\ dBw/m^2$	41 dBw	8 MHz
X-band	$-150\ dBw/m^2$	48 dBw	16 MHz
Ku-band	$-152\ dBw/m^2$	52 dBw	63 MHz

Table 1-9. System Frequency Plan

Links	Frequency	Channel Bandwidth
Forward Link		
LDR	400.5 to 401.5 MHz	1 MHz 4 - 250 kHz channels
MDR		
S-band	2025 to 2120 MHz	95 MHz channel
Ku-band	14.6 to 15.2 GHz	4 - 100 MHz channels
TDRS/GS		
Ku-band	13.4 to 13.64 GHz	240 MHz
VHF	148.26 MHz	
S-band	2200 to 2290 MHz	90 MHz
Return Link		
LDR	136 to 138 MHz	2 MHz (20 users multiple accessed/TDRS)
MDR		
S-band	2200 to 2300 MHz	20 - 10 MHz slots in 5 MHz steps or 100 MHz wide open
Ku-band	13.4 to 14.0 GHz	4 - 100 MHz channels
TDRS/GS		
Ku-band	14.6 to 15.2 GHz	200 or 600 MHz channels
VHF	136.11 MHz	--
S-band	2025 to 2110 MHz	85 MHz

1.3.2 Telecommunications Subsystem Design

To meet the design requirements and goals, the telecommunication subsystem analysis and design effort emphasized and stressed an overall design goal to maximize the simultaneous support of user spacecraft with the flexibility to adapt to anticipated changes in users needs. In addition, minimum size, weight, and power for the TDRS spacecraft as well as minimum impact on the user spacecraft terminals were emphasized. High reliability is achieved through full block redundancies of all critical components/subassemblies, functional alternative redundancies employing back-up via entire functional chain, and/or multiple operating channels which provide high reliability through graceful degradation. The recommended design reflects a state-of-the-art technology.

Figure 1-17 is a simplified block diagram of the telecommunications subsystem indicating the major elements and identifying the transceivers associated with supporting the 20 LDRUs and 2 MDRUs and maintaining the TDRS/GS interface.

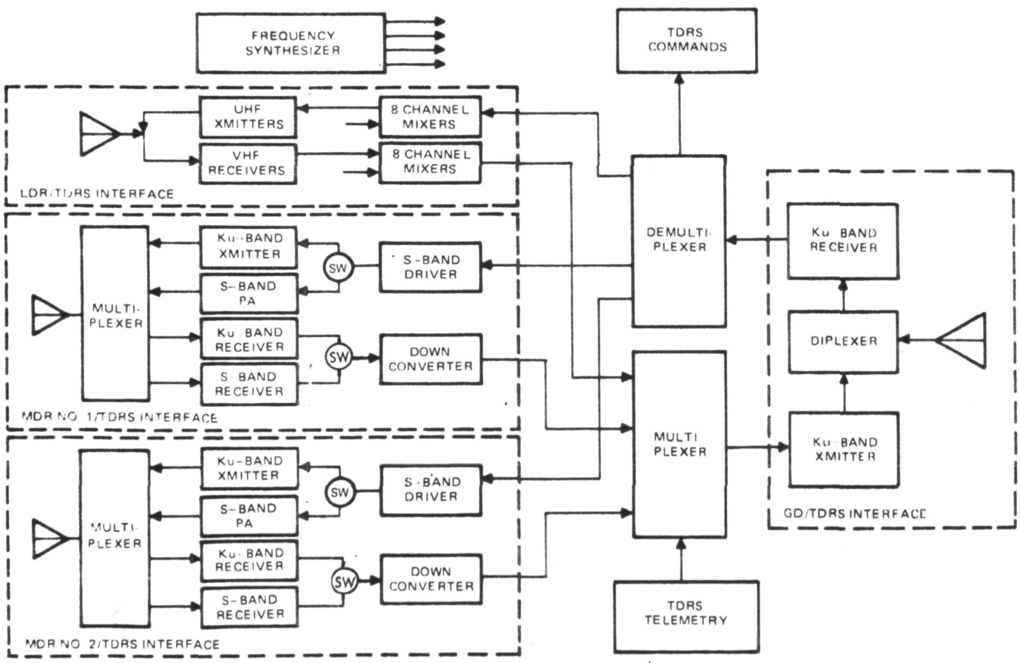


Figure 1-17. Telecommunications Subsystem Block Diagram

The alternative telecommunication subsystem operating modes are summarized in Table 1-10 along with the dc power requirements of each mode. The powers shown in the table do not include the effect of power conditioning.

Table 1-10. Telecommunications Power Requirements

		Operational Mode	dc Power Requirements
1	Return Link	20 LDRU (VHF) 2 MDRU (S-band)	256.0
	Forward Link	2 LDRU (UHF) 2 MDRU (S-band)	
2	Return Link	20 LDRU (VHF) 2 MDRU (1 S-band, 1 Ku-band)	224.1
	Forward Link	2 LDRU (UHF) 2 MDRU (1 S-band, 1 Ku-band)	
3	Return Link	20 LDRU (VHF) 2 MDRU (Ku-band)	191.1
	Forward Link	2 LDRU (UHF) 2 MDRU (Ku-band)	

1.3.2.1 LDR Transponder on TDRS

The TDRS satellite must provide a multiple access relay space-to-space interface with the low altitude users; and a space-to-ground interface with the ground station.

As previously indicated, the potential interference levels can totally disrupt the entire LDR link if adequate precaution is not taken during the early design phases.

In the LDR return link, the TDRS transponder sees nearly an entire hemisphere of RFI emitters, its level being potentially 20 to 40 dB greater than the desired user signal. To combat the high RFI environment the TDRS system uses an Adaptive Ground Implemented Phased Array (AGIPA) which adaptively employs spatial filtering and polarization discrimination of RFI emitters to maximize the signal-to-interference ratio (SIR). As the name AGIPA implies,

the spaceborne phased array conducts all of its beam steering/beam shaping and other signal processing functions at the ground station (GS). All the advantages of the phased array are achieved without placing the complexities of the multi-beam processing functions aboard the TDRS. AGIPA provides an independent beam and signal processor for each user, employing a mini-computer at the GS to provide the iterative SIR optimization.

A four-element phased array is used whose individual element characteristics are the same as a fixed field-of-view (F-FOV) approach which sees the entire 31 degrees (.54 rad) FOV.

The performance of AGIPA was compared with a F-FOV approach for the LDR return link. Since AGIPA is an adaptive system, a computer program and typical realistic RFI models were created to compare and assess the performances of the two approaches for typical user spacecraft orbits. For Atlantic and Pacific scenarios representing TDRS satellites at 11 degrees west and 141 degrees west longitude, AGIPA nominally provides 5 to 15 dB improvement in link performances as compared to a F-FOV approach. It also achieves the specification requirement of 10 kbps in relatively high RFI density. In contrast, a F-FOV approach is limited to less than 1000 bps and in the presence of a large jammer (e.g., 20 kw emitter), the F-FOV is totally inoperative. AGIPA discriminates the large emitter along with the other RFI emitters, thus optimizing the SIR.

The comparative analysis not only shows processing gain is necessary to achieve the specified link performance in the light of the expected RFI power density, but also indicates the shortcomings of the F-FOV to achieve the desired link performances and, more significantly, the potential to become totally disabled in the presence of a single large RFI emitter.

For the LDR return link, eight wideband (2.0 MHz) linear channels are used to relay the vertically and horizontally polarized components from the four AGIPA antenna elements. The loss of any channel or channels results in graceful degradation and merely represents reduced performance, thereby enhancing reliability.

The LDR forward link is also subject to RFI, but contrary to the return link, the user spacecraft receiver is confronted with the interference signals. Since user spacecraft can be more than 200 times closer to the RFI emitters than the TDRS, the TDRS transponder must overcome this large differential space loss to provide an adequate signal-to-noise ratio (SNR) at the user. This link was evaluated for operation at VHF, UHF, and S-band. The link analysis and TDRS hardware implementation impact support the recommendation to operate this link at UHF. The relatively large implementation impact at S-band eliminates it for this Part I phase. In comparing VHF and UHF in this RFI limited link, the link performances of both bands are equivalent. The 10 dB additional space loss at UHF becomes immaterial since the desired as well as the RFI signals suffer the same space loss. Although a conclusive definition of the RFI power level cannot be provided at VHF and UHF, it appears that the RFI power density exceeds the desired signal by approximately 30 to 40 dB; and the existing data indicate that the density is less at UHF. In addition, the effective isotropic radiated power (EIRP) is limited in both

VHF and UHF to +30 dBw to maintain the flux density below the IRAC established guidelines. As a consequence, the relative performance of these links can be evaluated for the same EIRP. By employing high gain backfire antenna elements colinearly stacked on each of the four VHF array elements, the hardware implementation tradeoff shows the UHF band can provide the maximum link performance for minimum size, weight, and power impact at the TDRS. The TDRS will form two beams on the satellite which will simultaneously support two command links to one or two independent LDR users, each with voice or data. Alternatively one beam can operate F-FOV to simultaneously illuminate all users at a reduced EIRP for coherent ranging or reduced data or voice capability.

1.3.2.2 MDRU Transponder on TDRS

In the MDR space-to-space link, considerable forethought and analysis were given to (1) insure flexibility for the transition between current and future needs, (2) provide operational flexibility to meet the frequency format of potential users, and (3) encompass the requirements of the TDRS specification and the Space Shuttle. Tradeoff analysis showed this link can meet the above needs with a dual frequency S-/Ku-band capability; S-band to meet the needs of current low performance ($H = 10^4 - 10^5$ bps) users, and Ku-band to meet the needs of the high performance ($H \geq 10^5$ bps) user including future TV requirement for the Space Shuttle. The TDRS provides two dual frequency MDR transponders to simultaneously support two independent users; both operating at S-band, both at Ku-band, or one at each frequency band. Link analyses show that the TDRS specifications as well as Space Shuttle requirements can both be supported with 6.5'(2 m) parabolic dishes. In the return link, both receivers were designed as a 10-MHz channelized receiver tunable in 20 discrete steps over the entire 100 MHz bandwidth; however, one of the receivers has the option to operate wide open with a 100 MHz bandwidth at S- and Ku-band to provide a true bent pipe repeater at the TDRS with growth capacity to handle TV. In addition, at Ku-band each receiver is tunable in four discrete steps over a 400 MHz bandwidth. This combination of a wide open/channelized receiver provides operational flexibility for the user signal format and frequency consistent with the weight and power limitations of the Delta 2914 launch vehicle.

In the forward link, each transmit channel is wideband with wide open 100 MHz channels at S-band, and 95 MHz channels tunable in four discrete steps over a 400 MHz bandwidth at Ku-band. One of the MDR antennas and/or transceivers can functionally provide a backup for the ground link at S-band or Ku-band.

1.3.2.3 TDRS/GS Transponder on TDRS

The Ku-band ground link provides the interface not only between the space-to-space links and the ground station, but also the ancillary functions that are necessary to support and service the TDRS satellite itself; viz. the coherent reference for the onboard frequency source, and tracking, telemetry, and command functions.

Tradeoff analysis of multiplexing approaches shows that a FDM/FM technique, trading power and weight for increased bandwidth, meets the goals of frequency flexibility, maximum link performance, and minimum size, weight, and power. To achieve the wide open operating flexibility in the MDR links, the Ku-band

frequency spectrum for the space-to-space and ground links overlap; however, because of the spread spectrum (FM) used in the space-to-ground link, this signal will be below the user receiver noise level and will not degrade its performance. This link operates with a 3'(0.9 m) dish on the TDRS and a 60' dish (18.3 m) at the ground station. The baseline design provides a rain margin of 17.5 dB. In addition, one of the two 6.5-foot (2 m) dishes can be used for this ground link to provide an additional 6.5 dB of gain, or a total of 24.0 dB of margin for operation in rain.

The on-board ground link transmitter output is adaptive, such that a 10-dB final RF power amplifier is inserted only for operation of this link in inclement weather. Consequently, the average power requirement can be maintained at a low 7.6 watts (dc) level for normal operation.

In the TDRS/GS forward link the ground station has additional flexibility to generate high RF power; consequently the on-board receiver is designed with a mixer front end. The RF front end is wide open (500 MHz wide) and subsequently down-converted and demultiplexed to provide the individual channelized data for relay to the LDR and MDR user spacecraft or for on-board coherent reference, tracking, telemetry, and command functions.

1.3.2.4 Frequency Synthesizer

A common frequency source locked to a pilot tone transmitted from the ground station generates the coherent signal references on the TDRS. A central frequency generator initially generates key reference signals which are subsequently used to lock-in remote voltage controlled oscillators (VCO). This approach minimizes the transmission of numerous RF reference signals throughout the on-board telecommunication system, allows each VCO to operate incoherently as a back-up mode, and minimizes the generation and filtering of spurious signals. In addition, the frequency source is fully redundant, and has additional back-up capability to become the prime reference source, so that the ground source can lock-in to the spaceborne reference.

1.3.2.5 TDRS TT&C and MDR Order Wire

An S-band transponder is used to provide trilateration on ranging, an MDR acquisition beacon, and an order wire for service.

For telemetry and command (T&C) functions, a digital approach with a central multiplexer/processor was selected. All command functions will be preprogrammed into a read-only memory device which can be commanded or scheduled to operate as required.

1.3.3 Telecommunications Relay Performance

The integrated attributes of the TDRS design and the telecommunications system design provide a very flexible and high capacity communication tracking and data acquisition support service. The design presented provides simultaneous support to 20 LDRUs and 2 MDRUs in the return link, 2 LDRUs and 2 MDRUs in the forward link, and simultaneous tracking of 20 users through each TDRS.

However, in the case of the LDRUs, even with a system that is optimized to combat interference, the service quality will be a function of the RFI **levels**; and in the case of the MDRUs the service quality will be a function of the characteristics of the user terminals. The quality of service (performance) is presented in the following sections in parametric form for each category of user spacecraft.

1.3.3.1 LDRU Forward and Return Link Performance

The forward link signal for LDR service must contend with multipath, RFI, and ambient noise at the user spacecraft. The impact of these effects can be minimized by obtaining the maximum processing gain through a specific bandwidth allocation. The processing gain (PG) obtainable within the system RF bandwidth constraints is proportional to the ratio of PN chip rate to the data rate. Forward link achievable data rates and rms range and range rate errors are shown in Figure 1-18.

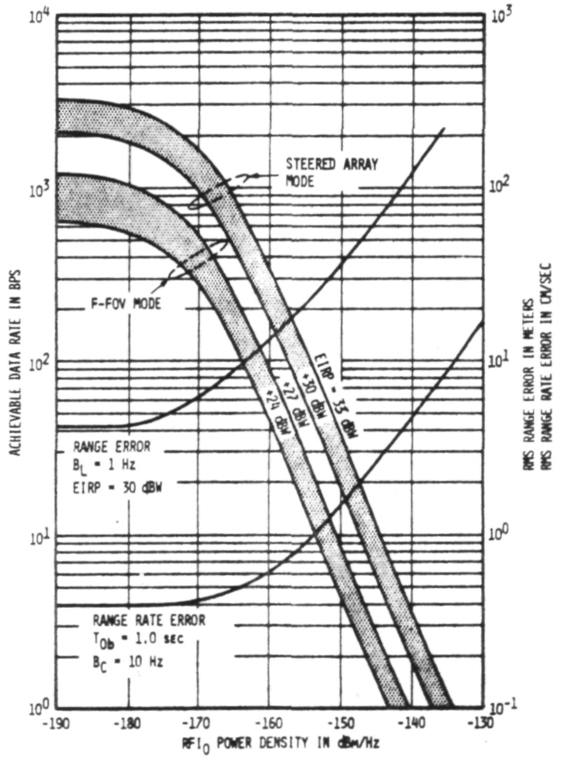


Figure 1-18. LDR Forward Link Performance
Achievable Data Rate, Range Error, and Range Rate Error

The return link signal through the TDRS must be capable of simultaneously supporting a maximum of 20 users, and the choice of PN rates, sequence lengths, and user data rates are interdependent.

The return link performance curves shown in Figure 1-19 are based on 5-watt (37 dBm) user transmit power (P_T) levels. There are two distinct regions of the curves; (1) a lower region where RFI limits performance, and (2) an upper region where performance is limited by noise, multipath, and other user signals. The curves are conservative in that all known potential losses have been included.

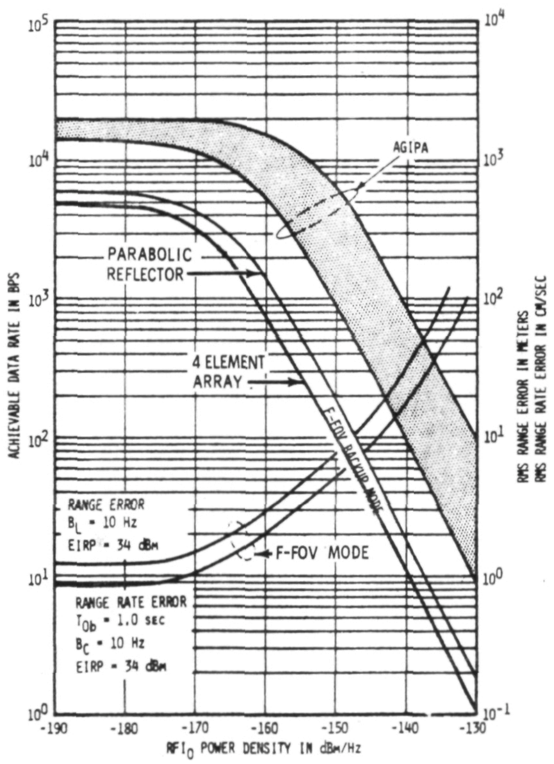


Figure 1-19. LDR Return Link Performance
Achievable Data Rate, Range Error, and Range Rate Error

Figure 1-19 includes forward error control (FEC) coding gain of 4.7 dB. Moreover, application of the AGIPA processing enhances system performance in the presence of RFI by an additional 5 to 15 dB as shown by the shaded area.

1.3.3.2 MDRU Forward and Return Link Performance

The links designed to support the MDRUs provide service at both S- and Ku-band. The capacity of these links depend on the characteristics of the user spacecraft terminal. Figure 1-20 shows the achievable data rate as a function of user EIRP for the return link (for both S- and Ku-band) and in Figure 1-21 for the forward link at S- and Ku-band as a function of user antenna gain. The curves include both specification and shuttle user requirements.

Tables 1-11 and 1-12 define the link characteristics and margins for specific channel capacities and user spacecraft characteristics for both the specified MDR requirements and for shuttle requirements at S-band.

Table 1-11. MDR Forward Link Performance

Item		MDR S-Band User		Manned User (Space Shuttle)		MDR Ku-Band User	
Modulation		ΔPSK		ΔPSK		ΔPSK	
Transmitter power	dBw	12.5		12.5/18.5		-2.0	
TDRS antenna gain	dB	29.8		29.8		48.1	
Transmit line losses	dB	1.3		1.3		1.0	
EIRP	dBw	41.0		41.0/47.0[7]		45.1	
Losses							
Space	dB	-192.0		-192.0		-208.1	
Pointing	dB	-0.1		-0.1		-0.1	
Polarization	dB	-0.5		-0.5		-0.5	
User antenna gain	dBi	G_u		3.0		G_u	
Received power	dBw	$-151.6 + G_u$		-148.6/-142.6		$-163.6 + G_u$	
System noise temperature	dB	29.1		27.3[1]		33.5[5]	
Thermal noise density	dBw/Hz	-199.5		-201.3		-195.1	
TDRS ΔCNR degradation	dB	-0.25		-0.25		-0.25	
Available C/N_0	dB-Hz	$47.65 + G_u$		52.45/58.45		$31.25 + G_u$[6]	
Support Requirements		100 bps[2]	1000 bps	Data[3]	Data + 1 Voice[4]	Data + 2 Voice[4]	BER = 10^{-5} (ΔPSK)
C/N_0 required	dB-Hz	29.9	39.9	41.7	49.6	52.4	--
Data rate achievable	bps	--	--	--	--	--	$68.4 \times G_u$[6]
Design margin	dB	$17.75 + G_u$	$7.75 + G_u$	10.75	8.85	6.05	3.0

NOTES
1. Noise temperature with an uncooled paramp.
2. For ΔPSK with a BER = 10^{-5} (E_b/N_0 = 9.9 dB)
3. Data = 2 kbps ΔPSK with BER = 10^{-4} (E_b/N_0 = 8.7 dB)
4. Voice is delta modulated at 19.2 kbps; carrier modulation is ΔPSK (BER = 10^{-3}; (C/N_0 = 48.8 dB-Hz)
5. Tunnel diode amplifier
6. A 1.0 ft (.3 m) antenna at Ku-band provides about 28.5 dB gain (i.e., a factor of about 800) providing an achievable bit rate of about 55 kbps
7. Increased EIRP used for voice mode only

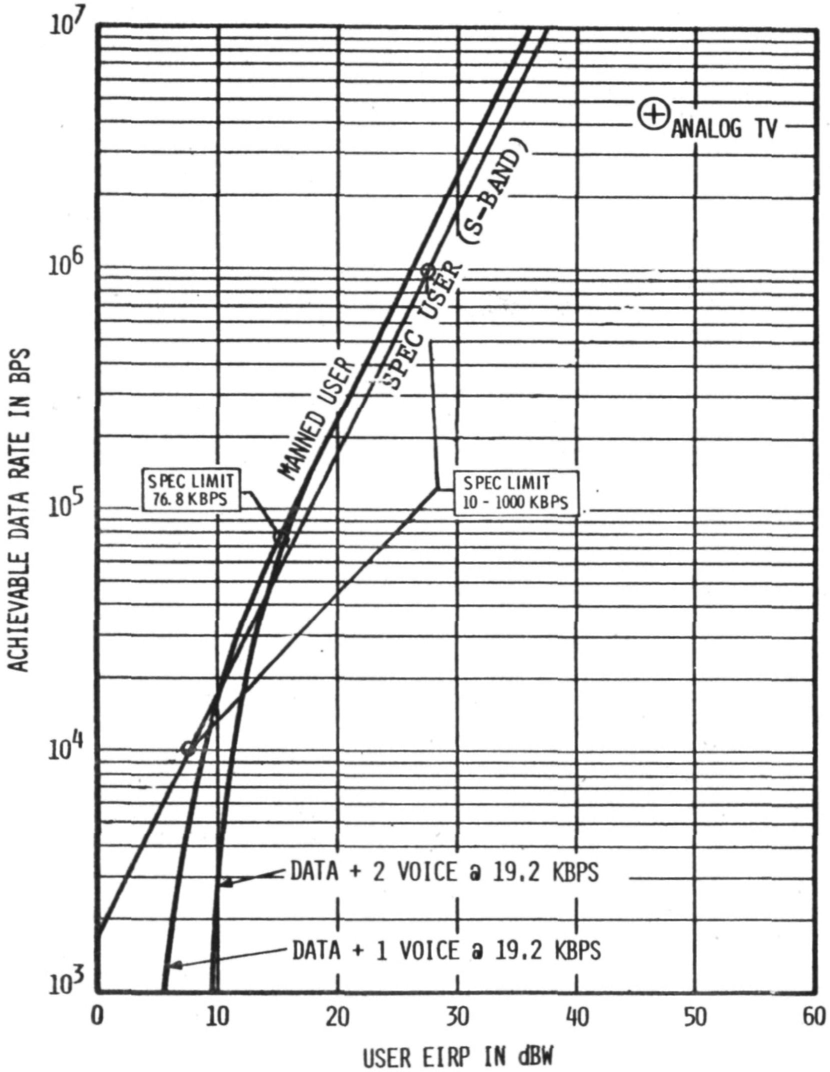


Figure 1-20. MDR Return Link Performance

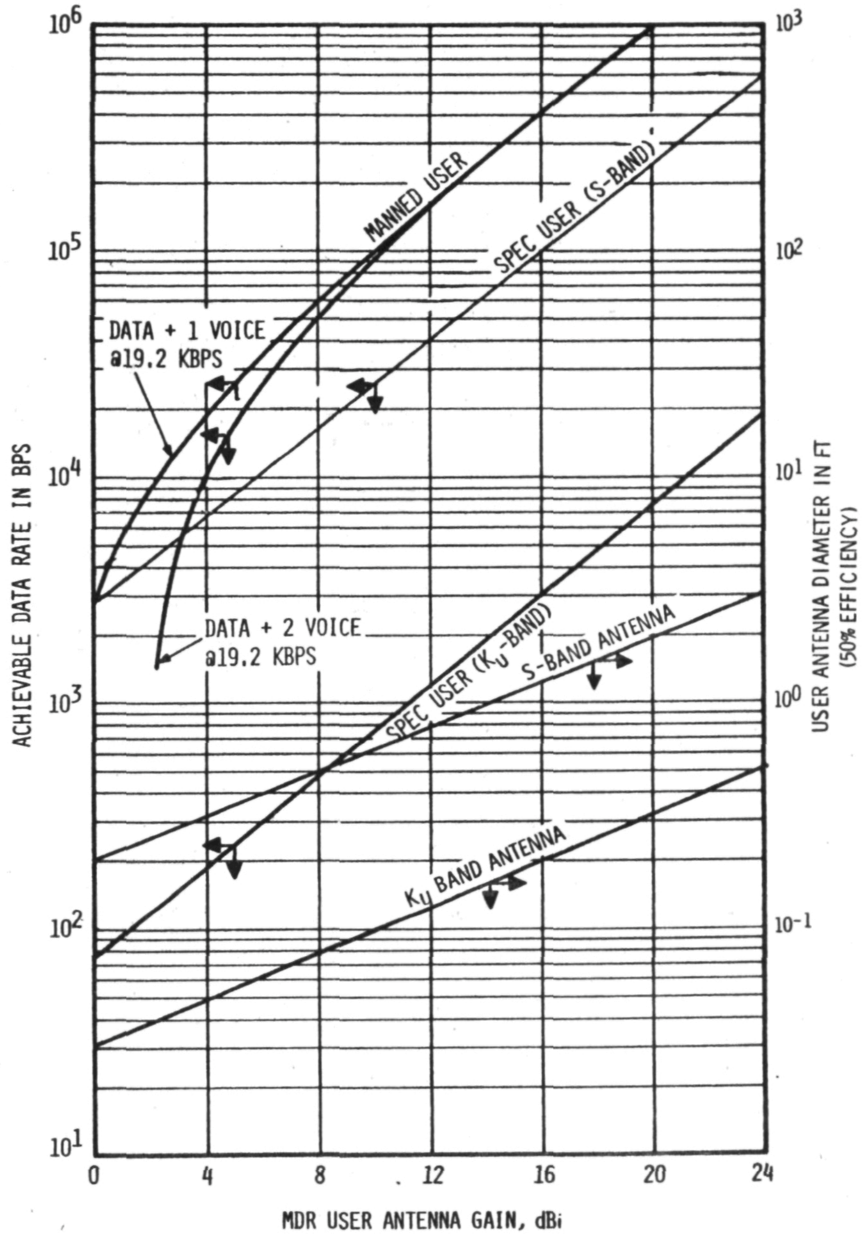


Figure 1-21. MDR Forward Link Performance

Table 1-12. MDR Return Link Budget

(a) S-Band

Item	MDR User			Manned User (Space Shuttle)			
	10 kbps[1]	1000 kbps[1]	Analog TV[2]	Data[3] (76.8 kbps)	Data + 1 Voice[4]	Data + 2 Voice	Analog TV
C/N_0 required, dB-Hz	49.9	69.9	87.54	57.59	58.17[5]	58.50[6]	87.54
FEC coding gain, dB	4.70	4.70		4.70	4.70	4.70	
Effective C/N_0, dB-Hz	45.2	65.2	87.54	52.89	53.47	53.80	87.54
Path loss[7], dB	191.1	191.1	191.1	191.1	191.1	191.1	191.1
System temperature[7], dB	26.2	26.2	26.2	26.2	26.2	26.2	26.2
Boltzmann's const, dBW/°K-Hz	-228.6	-228.6	-228.6	-288.6	-228.6	-228.6	-228.6
Pointing loss, dB	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Polarization loss, dB	0.5	0.5	0.5	0.5	0.5	0.5	0.5
G_{user}, dBi	G_u	G_u	G_u	3.0	3.0	3.0	3.0
G_{TDRS}, dBi	30.9	30.9	30.9	30.9	30.9	30.9	30.9
ΔCNR degradation, dB	1.0	1.0	1.0	0.5	0.5	0.5	0.5
System margin, dB	3.0	3.0	3.0	3.0	3.0	3.0	3.0
Required EIRP, dBw	7.6	27.6	49.94	14.79	15.37	15.70	49.44
Transmit power[8], dBw	$7.6 - G_u$	$27.6 - G_u$	$49.94 - G_u$	11.79	12.37	12.7	46.44

NOTES
1 ΔPSK; BER = 10^{-5} (E_b/N_0 = 9.9 dB)
2 Commercial quality $(S/N)_o$ = 40 dB; BW = 4.5 MHz
3 ΔPSK; BER = 10^{-4} (E_b/N_0 = 8.7 dB)
4 Voice is delta modulated at 19.2 kbps; carrier modulation is ΔPSK (BER = 10^{-3}; C/N_0 = 48.8 dB-Hz)
5 Combine value of data (C/N_0 = 57.59) and one voice (C/N_0 = 48.8)
6 Combine value of data (C/N_0 = 57.59) and two voice (C/N_0 = 51.8)
7 This value is selected at the point where the product of the path loss and system temperature is maximum
8 Transmit power into antenna

(b) Ku-Band

Item	10 kbps[1]	1000 kbps[1]	Analog TV[2]
C/N_0 required, dB - Hz	49.9	69.9	87.54
FEC coding gain, dB	4.7	4.7	-
Effective C/N_0, dB - Hz	45.2	65.2	87.54
Path loss[3], dB	208.1	208.1	208.1
System temperature[3], dB	26.7	26.7	26.7
Boltzmann's const., dBw/°K - Hz	-228.6	-228.6	-228.6
Pointing loss, dB	0.1	0.1	0.1
G_{user}, dB	G_u	G_u	G_u
G_{TDRS}, dB	48.3	48.3	48.3
ΔCNR degradation, dB	0.5	0.5	0.5
System margin	3.0	3.0	3.0
Required EIRP, dBw	6.7	26.7	49.04

1 ΔPSK; BER = 10^{-5} (E_b/N_0 = 9.9 dB)
2 Commercial quality $(S/N)_0$ = 40 dB; BW = 4.5 MHz
3 This value is selected at the point where the product of path loss and system temperature is maximum.

1.4 SPACECRAFT MECHANICAL AND STRUCTURAL DESIGN

The guidelines during the synthesis of design concepts and the criteria for decision making, for both design variations and for selection between alternative configurations, was maximum relay support capability and maximum telecommunications flexibility within the constraint of providing a low cost, low risk design, launched on a Delta 2914.

The evolution of the baseline spacecraft design proceeded as illustrated in Figure 1-22, and within the constraints of Table 1-13. Although the problems addressed during Part 1 of this study were extremely challenging, the resulting spacecraft concept is simple and straightforward.

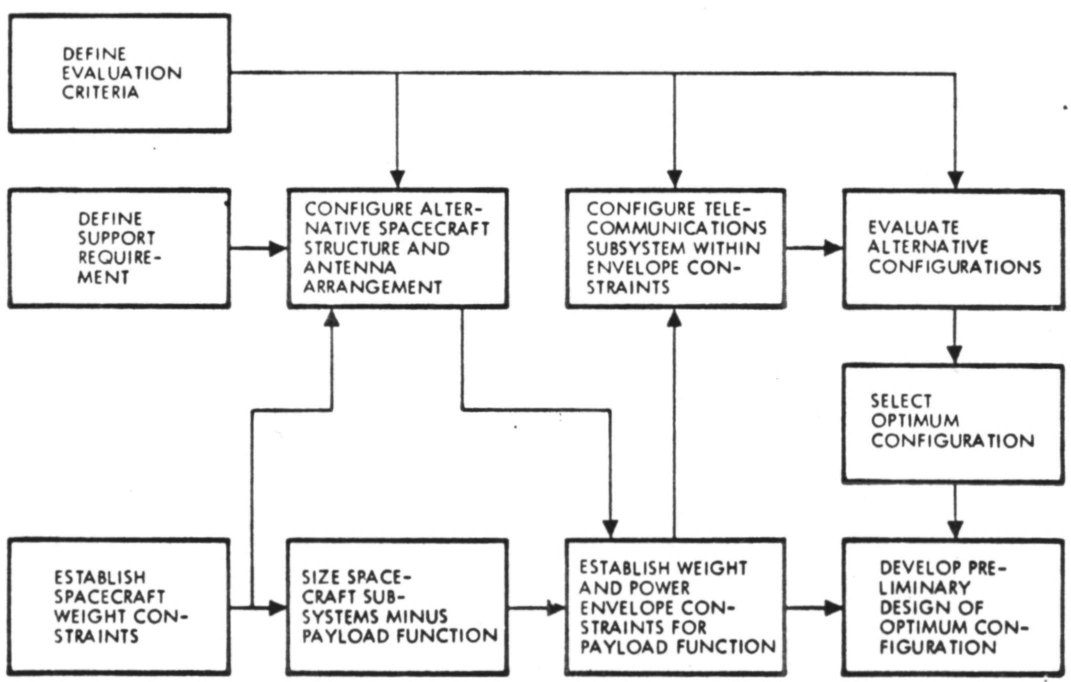


Figure 1-22. Evolution of Spacecraft Design

Table 1-13. Design Constraints

Constraint	Derived From
Maximum payload: 788 lb (357.5 kg) (Including burned-out apogee motor case)	Delta 2914 capability
Volume	Delta 8-foot (2.4 m) shroud
No shadowing on solar cells	Power loss cannot be tolerated
RCS system temperature >40 F (4.4 C)	Hydrazine freezes
Electronics temperature < 40 C	Reliability
RCS jets operable in a stowed configuration	Stability and pointing requirements in transfer orbit
Unfurlable antenna dishes not desired	Reliability, surface control, and minimum cost
Clear sensor FOV before and after deployment	Stability and pointing requirements

The spacecraft provides the support for, integrates and protects the communications, electrical power, attitude control, and propulsion subsystems in both the launch and deployed environments. The deployable elements are packaged to withstand the structural and vibrational loads experienced during launch on the Delta 2914 booster and the spinning accelerations imposed during third stage burn and transfer orbit. After apogee motor burnout, despin and stabilization maneuvers, the antennas and solar panels are deployed to their extended positions.

Areas that were of primary concern in the development of the final baseline design were packaging and deployment mechanization, antenna design and sizing, weight control, and optimization of the configuration for general-purpose support of multiple user spacecraft.

Figure 1-23 illustrates the arrangement of antennas and solar array panels symmetrically grouped around the central spacecraft body. The two MDR parabolic reflector antennas are supported on struts on each side of the LDR UHF-VHF four-element array of backfire antennas with the body-mounted TDRS/GS antenna in the center. The one-degree-of-freedom solar panels are deployed above and below the spacecraft beyond the solar shadow limits of the antennas. Backup TT&C omni whip antennas located around the rear of the spacecraft are utilized during launch and spacecraft orbital maneuvers prior to the deployment of the primary antennas.

Figure 1-23. TDRS Baseline Configuration

The spacecraft is packaged for launch within the 8' (2.4 m) shroud of the Delta 2914. The MDR antennas are folded forward in a face-to-face position with slight mutual rotation to allow the antenna feeds and supports to clear each other. The 3-ft dia (0.9 m) TDRS/GS dish antenna is located at the front of the spacecraft between the rear rims of the angled MDR dishes. The LDR elements are packaged into four cylindrical shapes and positioned around the TDRS/GS antenna behind the MDR dishes forward of the structural body of the spacecraft. The solar array panels are folded down around the sides of the spacecraft allowing room for clearance with the side-mounted booms of the MDR antennas and the ACS thrusters.

The spacecraft achieves its deployed configuration by extending the solar arrays, and the two MDR antennas aft to their positions on each side of the spacecraft, and deploying the four elements of the LDR UHF-VHF array laterally and then extending them forward. The LDR antenna is deployed laterally by spring-loaded linkage joints which lock into the extended position. Release of the deployable elements is initiated by solenoid activation of packaging and restraint latches by ground commands. After the lateral deployment the LDR antenna elements are extended forward by motor driven "STEM" units mounted at the rear of the elements and the extended "STEM's" become the center rod supports for the disc elements.

The spacecraft body (Figure 1-24) consists of inner aluminum tapered cones around the apogee motor, a transverse equipment shelf of aluminum honeycomb and the outer body shells of aluminum honeycomb that close off and protect the internal equipment and house the thermal louvers.

The apogee motor is installed on the spacecraft centerline by an accurately machined titanium ring attached to the apogee motor flange and is bolted to the inner structural cone. The attachment ring is located along the Z axis of the spacecraft to accurately position the apogee motor. The motor nozzle extends 2.5"(6.4cm) beyond the spacecraft separation plane. The apogee motor is not jettisoned after burnout and remains in the spacecraft body.

The equipment shelf is an aluminum honeycomb bulkhead that provides the primary mounting surface for the equipment. It is fabricated of 1.50-in.(3.8 cm) thick aluminum honeycomb core with 0.010"(.025 cm) alum. face sheets. The insert panels are of similar construction and bolt to the main bulkhead. Equipment is mounted on both the bulkhead and the insert panels, allowing for simultaneous and sequential subsystem installation and checkout during manufacture.

The outer body shells are bonded 0.50" (1.3 cm) thick alum. honeycomb core with 0.010" (.025 cm) reinforced fiberglass face sheets. The shell halves are bolted to the inner shell structure and the outer rim of the equipment shelf using angle clips.

The two MDR antennas are dual Ku- and S-band frequency, 6.5' (2 m) diameter, solid-face, parabolic reflectors with two-axis gimbal drives. They are mounted on tubular booms and deployed 138.50"(3.52 m)outboard on each side of the spacecraft. As shown in Figure 1-23, the packaging restraints of the 8' (2.4 m) shroud in conjunction with the volume required for the spacecraft body, solar arrays, and packaged UHF-VHF array elements, limit the two solid-face antennas

to a maximum of 6.5'(2 m) diameter. Solid-face reflectors were chosen over furlable antennas because they meet all ERP and G/T requirements, and for simplicity of design, accuracy of surface tolerances suitable for Ku-band antenna frequencies, weight, rigidity and reliability.

The LDR antenna is an array of dual frequency backfire elements spaced uniformly around the spacecraft centerline. Each element consists of a series of discs-on-rod, a UHF dipole and mesh ground plane, and a VHF dipole and mesh ground plane. When deployed, each element is supported and positioned by a swing arm support link that places the element centerline at 48.75" (1.24 m) from the X and Y axes.

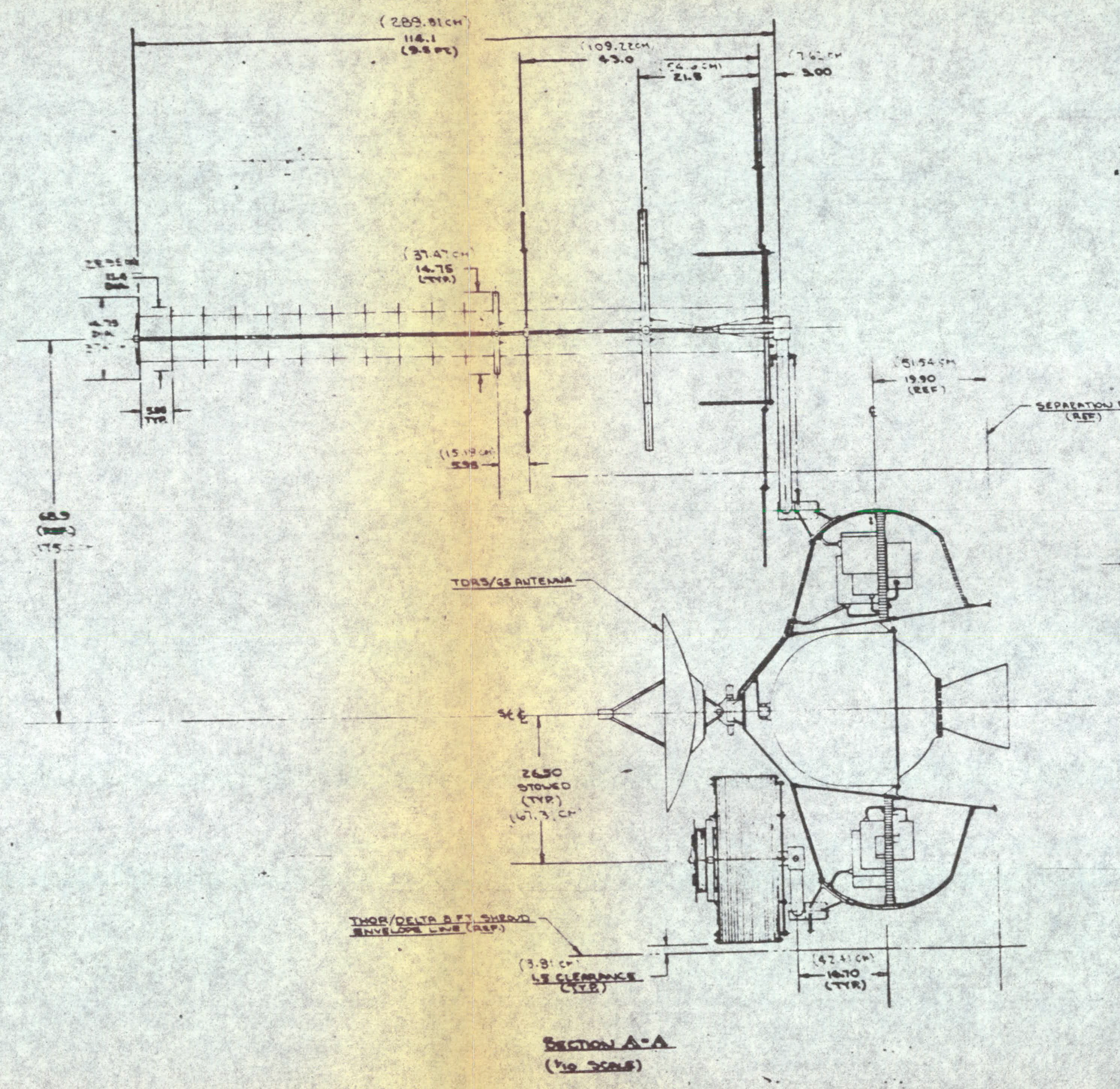
To package the UHF-VHF array elements in the available space with the other antennas and spacecraft body, each array element must be packaged in a volume 30" (76 cm) in diameter by 16" (40.6 cm) long, and must be deployed with a simple, lightweight, reliable arrangement with no loose packaging parts. The mechanical design of these elements is illustrated in Figure 1-25. The two large-diameter VHF ground planes fold down on spring-loaded arms which are equally spaced around a central mesh-frame disc 30" (76 cm) diameter. The VHF dipole assembly is reduced to the 30" (76 cm) diameter by compressing the spring-loaded dipole extensions. The discs, the UHF dipole, and ground plane, and VHF dipole are closely stacked against the front of the STEM actuator. The mesh rims of the two ground planes are folded into the space between the two ground plane discs in their stacked position. The ground plane arms are spring-loaded and restrained by lock pin shafts that extend forward from a locking disc behind the VHF ground plane. To release the mesh arms and allow the ground plane meshes to extend to their full diameters, this locking disc is rotated on the case of the STEM by a cable system activated by the lateral deployment of the support struts. This small rotation releases the mesh arms from the lock pin shafts and extends the mesh surfaces.

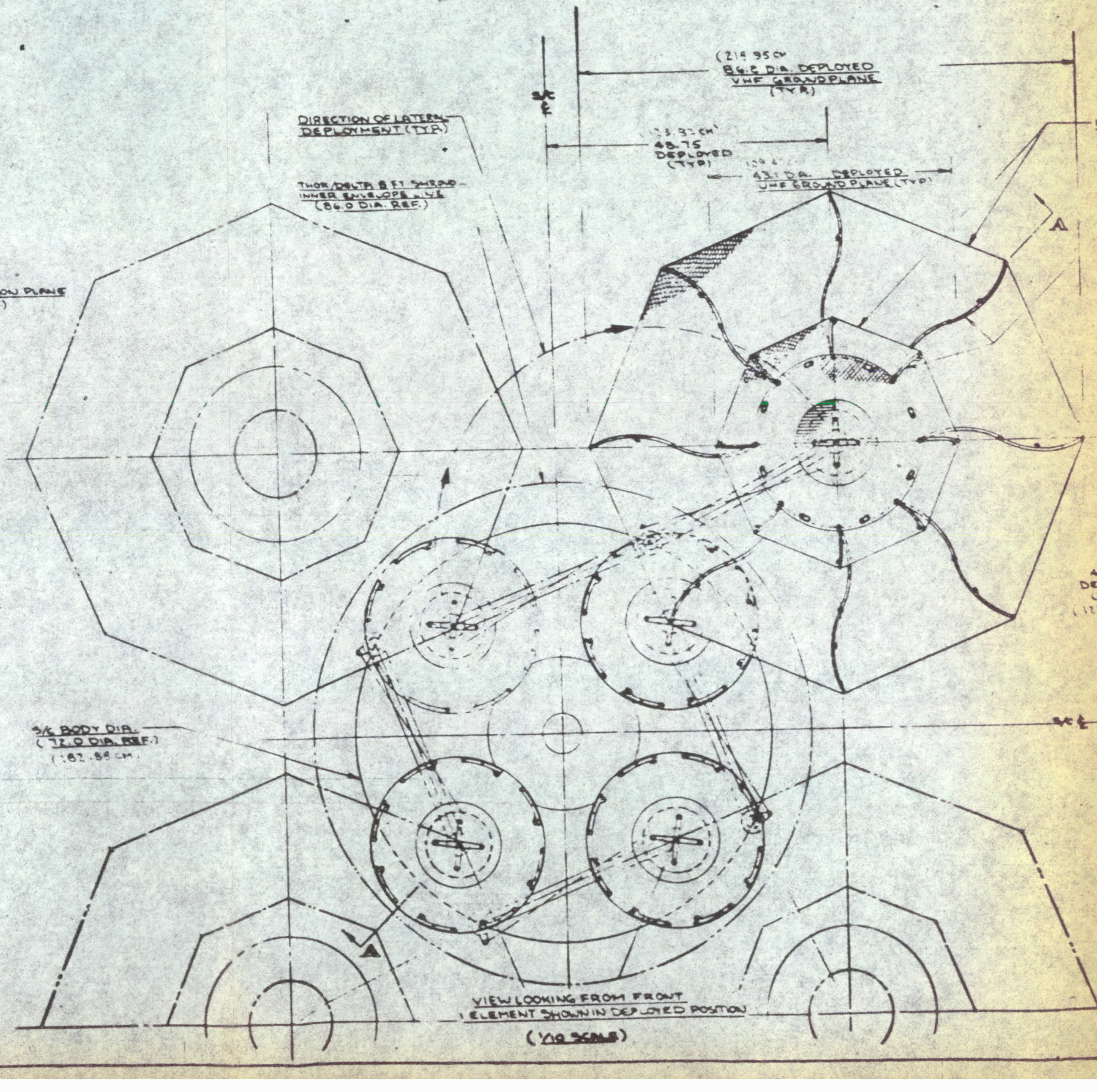
The ground link antenna is a 3-ft-diameter (.9 m) Ku-band parabolic reflector located on the centerline of the spacecraft, forward of spacecraft body. To provide clearance without blockage or reflective problems for the antenna beam between the LDR array elements, the antenna is positioned fore and aft as shown in Figure 1-23.

The tracking, telemetry and command backup antennas are VHF omni whips. One set of four whips located radially around the rear of the spacecraft is utilized during launch when the primary antennas are in their stowed position. These TT&C antennas are behind the stowed solar panels and after the shroud is jettisoned they have a clear field of view to the ground tracking stations. After three-axis stabilization and deployment of the primary antennas, a TT&C backup to the TDRS/GS Ku-band link is supplied by another set of the VHF omni-whip antennas mounted around the rim of the TDRS/GS Ku-band antenna.

Ku- and S-band tracking beacon antennas are mounted on the forward side of the spacecraft body adjacent to and offset from the TDRS/GS antenna. The Ku-band acquisition beacon antenna consists of a section of waveguide which terminates into a conical horn with a peak gain of nominally 12 dB and a 31° (.54 rad) FOV. The S-band acquisition beacon/order wire antenna is a 2" (.5 cm) diameter helix mounted on an 11" (28 cm) ground plane to provide performance similar to the Ku-band beacon antenna.

Figure 1-24. Spacecraft Body Configuration





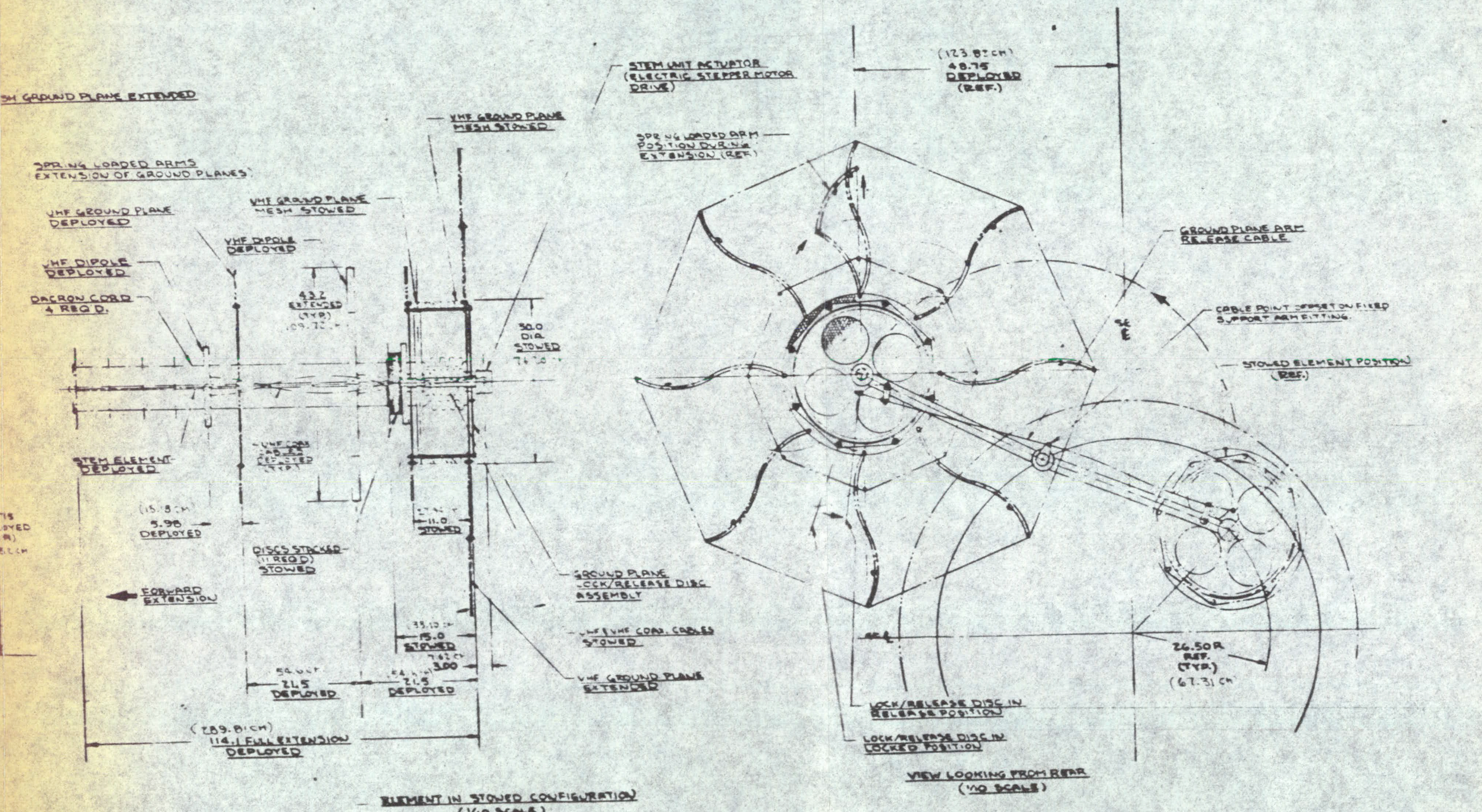


Figure 1-25. UHF-VHF Backfire Array Element

The solar array panels are deployed to the positions shown in Figures 1-23 and 1-26, and rotate at one revolution in 24 hours to maintain solar illumination on the front of the array.

To package the solar array in the launch configuration, the panels are constructed on a radius that permits the panels to fold around the top and bottom of the body. The gaps between the panels permit clearance for the MDR antenna booms and operation of the ACS thrusters in the launch configuration. The solar panel support links fold forward and under the solar panels and are restrained during launch loading by solenoid operated latch-release mechanisms. In this position the solar panels are active and provide limited electrical power during the spinning sequence.

After stabilization of the spacecraft at synchronous orbit, ground commands activate solenoids to release the linkage. Spring-loaded fittings extend the solar panels and lock the joints. As the panels extend, spring-loaded hinges, between the panel halves and the support link, which are also released by the solenoid mechanism, extend the solar panel halves forward from their circular stowed configuration to provide a flatter shape for increased efficiency. Further flattening of the panels would result in undue complexity since satisfactory performance is provided by this design. After full panel deployment, the panel drive actuators at the base of the linkage system orient the solar panel sun sensors with the sun and begin the 1 rev/day to maintain solar alignment.

The drive system, consisting of two actuators driven by electric stepper motors mounted on the rear surface of the equipment shelf, supports and rotates the deployed panels. Section A-A, on Figure 1-25, illustrates details of the Spar Aerospace Products actuator.

Most of the satellite equipment is installed on the front and back faces of the equipment shelf to provide proper center-of-gravity location, eliminate center-of-gravity travel during use of expendables, and provide ease of assembly, servicing and checkout. Figures 1-27 and 1-28 illustrate front and rear views of the equipment bulkhead with the subsystems installed.

After assembly and checkout of the propulsion system on the shelf, the attitude control system is installed and checked out. The communication system components are mounted on honeycomb panel inserts which match the equipment shelf cutouts, and are interconnected for bench checkout and testing as a subsystem unit. The communications panel inserts are then attached to the equipment shelf and the electrical power subsystem is installed on the rear face of the shelf and all cabling installed and tested.

The spacecraft equipment located on the equipment shelf provides convenient servicing through access panels on the front and rear outer body shells. At initial installation, the subsystems are mounted on the shelf prior to the assembly of the outer body shells and are fully exposed for checkout. Propulsion system fill and drain valves are accessible and convenient with the spacecraft in the stowed configuration. All launch restraints and solenoid release mechanisms are visible for prelaunch checkout and inspection. Prelaunch electrical check points are adjacent to the MDR antenna attachment fittings and

are readily accessible. Access to the Delta third-stage motor through the existing access holes in the attach fitting remains clear and convenient.

Access to and visual inspection of the mating Delta clamp securing the TDRS to the attach fitting is clear of structure and convenient for mating prior to launch.

The weight summary for the TDRS is shown in Table 1-14. Preliminary center-of-gravity location and moments of inertia for both the launch and deployed configurations are summarized in Table 1-15.

Table 1.14. TDRS Weight Summary

	Weight (lb)	Weight (kg)
Communications		
Electronics	122.2	55.5
Antennas	117.9	53.5
Attitude stabilization and control	57.7	26.2
Electric power	97.0	44.0
Solar array	58.6	26.6
Structure	91.0	41.3
Thermal control	23.9	10.8
Auxiliary propulsion hardware	38.4	17.4
	606.7	275.0
Propellant + N_2 (2-65° - 15-day station changes)	49.3	22.4
Total spacecraft	656.0	297.6
Contingency	82.0	37.2
Allowable P/L (Delta 2914 + CTS apogee motor)	738.0*	334.8*
Empty apogee motor case	50.0	22.7
Initial on-orbit	788.0	357.5
Burned-out insulation	8.0	3.6
Apogee motor propellant	688.0*	312.1
Synchronous orbit injection	1484.0	673.2
Transfer orbit propellant	6.0	2.7
Delta separation weight (27° transfer orbit)	1490.0	675.9
*5 deg/day drift orbit		

Figure 1-26. Solar Panel Array

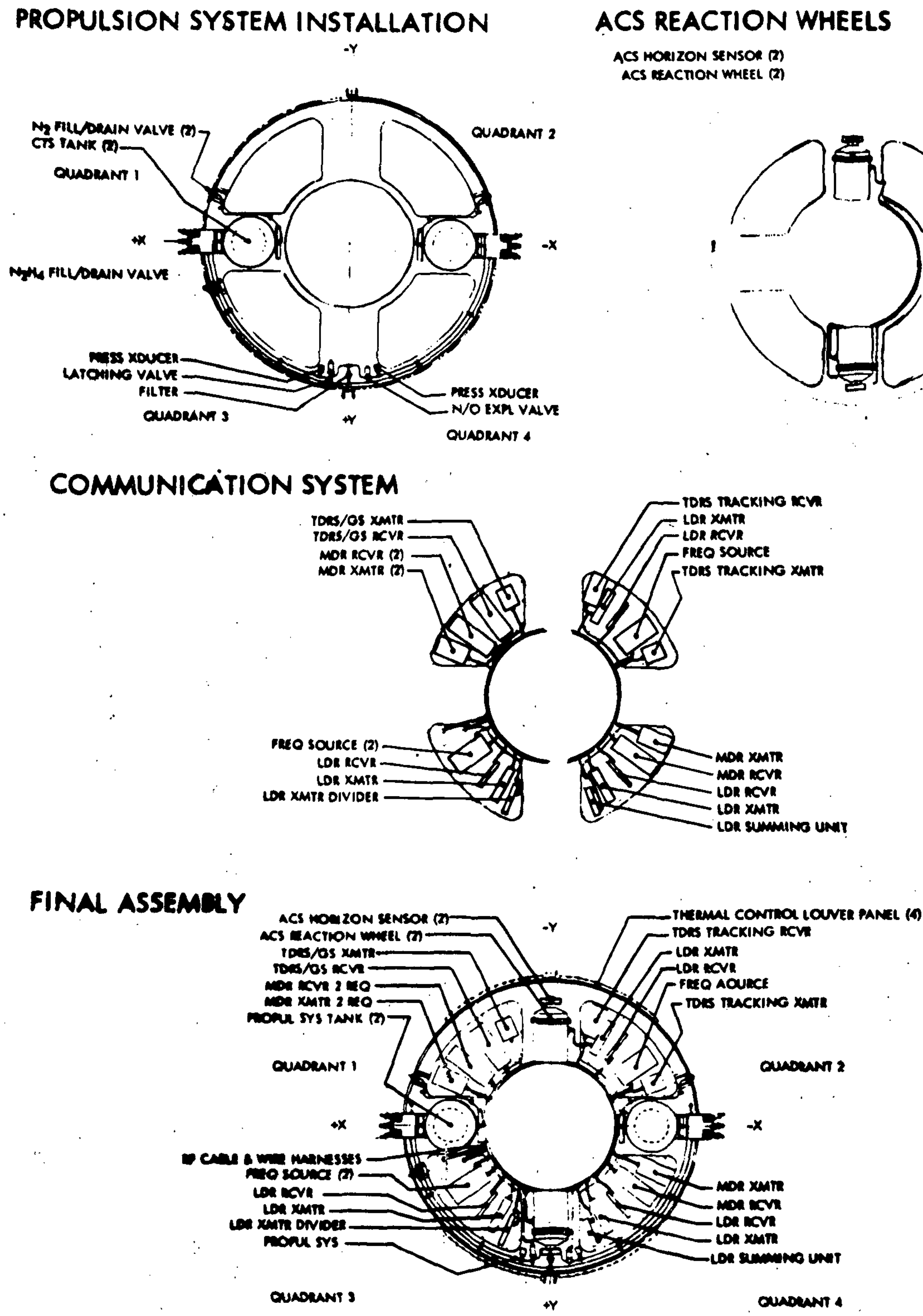


Figure 1-27. Equipment Shelf - Front View

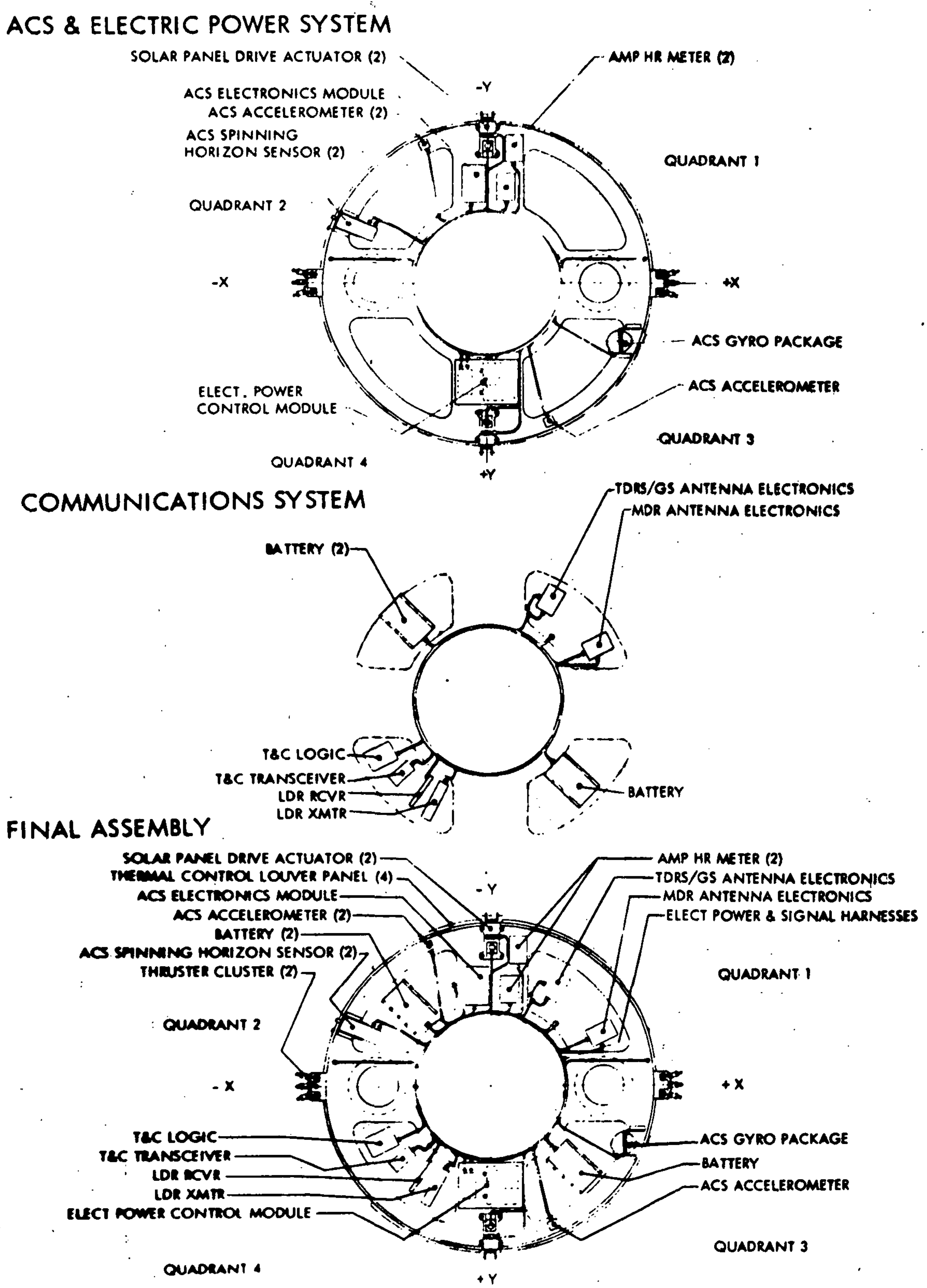


Figure 1-28. Equipment Shelf - Rear View

Table 1-15. Preliminary Moments of Inertia

Configuration	Weight		Z Center-of-Gravity (from S.P.)		Inertia					
					I_{z-z}		I_{x-x}		I_{y-y}	
	lb	kg	in.	cm	slug-ft^2	kg-m^2	slug-ft^2	kg-m^2	slug-ft^2	kg-m^2
Launch										
Full apogee motor	1484	673.2	25.7	65.3	107	145	132	179	122	165
Burnout apogee motor	788	357.5	26.1	66.3	94	127	119	161	108	146
Deployed										
Full propellant	788	357.5	21.6	54.7	465	630	234	317	308	418
Fuel expelled	738.7	335.0	21.6	54.7	458	621	234	317	301	408

*x and y center of gravity, 0.00 and 0.00

1.5 ELECTRICAL POWER SUBSYSTEM

Design of the electrical power subsystem was accomplished with consideration for five-year lifetime, eclipse operations, sun angle, battery redundancy, excess power dissipation, and energy transfer mechanisms. A primary design goal was to use only flight-proven technology and hardware. As a result, nickel-cadmium batteries were selected for the energy storage assembly. The selection of two batteries for the baseline was made after examining the impact of a battery failure and considering system weight. The loss of one of the two batteries still permits an eclipse load of 195 watts for the full 72 minutes which is sufficient to operate partially one forward LDR and one forward MDR link. A direct energy transfer concept was selected since this permits power transfer directly from the solar array to the loads without any in-line power conditioning. The majority of time (all but 80.2 hours per year) is spent in direct sunlight with spacecraft loads supported directly from the solar array, thereby minimizing power conditioning losses.

During transfer orbit an average power requirement of 44 watts must be provided. The baseline design utilizes the solar arrays in the stowed configuration. This is done by curving the panels and exposing the cells to simulate a body-mounted panel. The dependence on batteries is minimized by providing a projected area equivalent to 44 watts output for the minimum allowable sunline/spacecraft orientations.

The EOS power requirements of 300 watts (daylight average) and 213 watts (eclipse average) are shown in Table 1-16, and the functional block diagram in Figure 1-29. Power is supplied to the loads from a central regulated 28 $\pm$1 volt dc bus. Voltage is regulated by a shunt regulator operating as a variable load across lower sections of the solar array panels. Each panel is approximately 22.5 ft^2 (2.09 m^2) in projected area for a total area of 45 ft^2 (4.18 m^2). This area provides a beginning of life power of 466 watts, and an end of life power of 400 watts. This can be readily increased by 25 percent by extending the solar panels 10 inches (25.4 cm) below the booster separation plane.

Table 1-16. Electrical Power Requirements

SUBSYSTEM	DAYLIGHT AVE	DAYLIGHT PEAK	ECLIPSE AVE	ECLIPSE PEAK	TRANSFER ORBIT
ATTITUDE STAB. & CONTROL	16.5	100.5	13.5	85.5	5.2
HEATERS	2.0	25.2	1.0	25.2	10.4
TT&C	10.5	15.0	10.5	15.0	14.5
TELECOMM. SERVICE	249.3	344.3	179.4	274.5	4.8
EPS (ARRAY DRIVE AND CONTROLS)	15.7	33.2	8.5	33.2	5.2
CONTINGENCY	6.0		0		
SUBTOTAL	300.0		213.0		40.2
BATTERY CHARGE	48.0				
POWER CONDIT. LINE LOSSES	52.0		17.0		3.9
ARRAY OUTPUT E.O.L.	400.0				
DEGRAD ALLOW (5 YRS)	66.0				
ARRAY OUTPUT B.O.L. (45 FT^2)	466.0				
BATTERY LOAD			230.0		44.0

BASELINE:
NORMAL DAYLIGHT 2 LDR (ONE VOICE 25% DUTY CYCLE)
2 MDR (1S + 1 KU BANDS)
ECLIPSE 1 LDR (VOICE 25% DUTY CYCLE) 1 MDR (S-BAND)

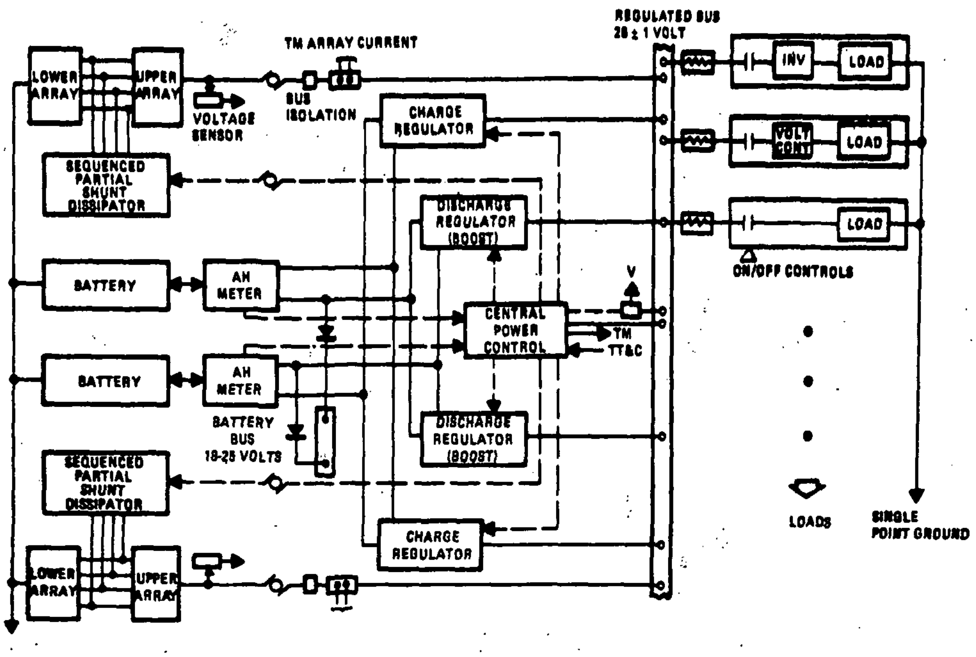


Figure 1-29. Electrical Power Subsystem Block Diagram

The central power control unit controls the various EPS operational modes. It detects the differences between the main bus and reference voltage levels. Since the solar array is a constant power source, to supply larger amounts of power to subsystem loads, battery charging must be inhibited and/or the boost regulator activated to supply power from the batteries. To provide for sun eclipse energy demands, a battery charging allowance of 48 watts (net to the batteries) is included in the sizing model to permit parallel battery charging.

Figure 1-30 shows a typical battery charging time for two NiCd batteries. Several charge characteristics will be available to permit more rapid battery charging depending upon availability of solar array power. A weight summary of the EPS is shown in Table 1-17. The voltage converter is required to provide 18 volts to the LDR for data transmission. This converter operates from the 28-volt regulated bus. As a general utility service the EPS delivers regulated 28-volt dc power with the exception of 18 volts dc for the LDR transmitters, all other nonstandard power conditioning will be part of the user equipment.

The beginning of life design is capable of supporting the higher requirements for two S-band by utilizing the contingency (6 watts) and the solary array degradation allowance (66 watts). Dependence on S-band is expected to decline with mission time. The end-of-life design will support one S-band and one Ku-band or 2 S-band without emergency voice.

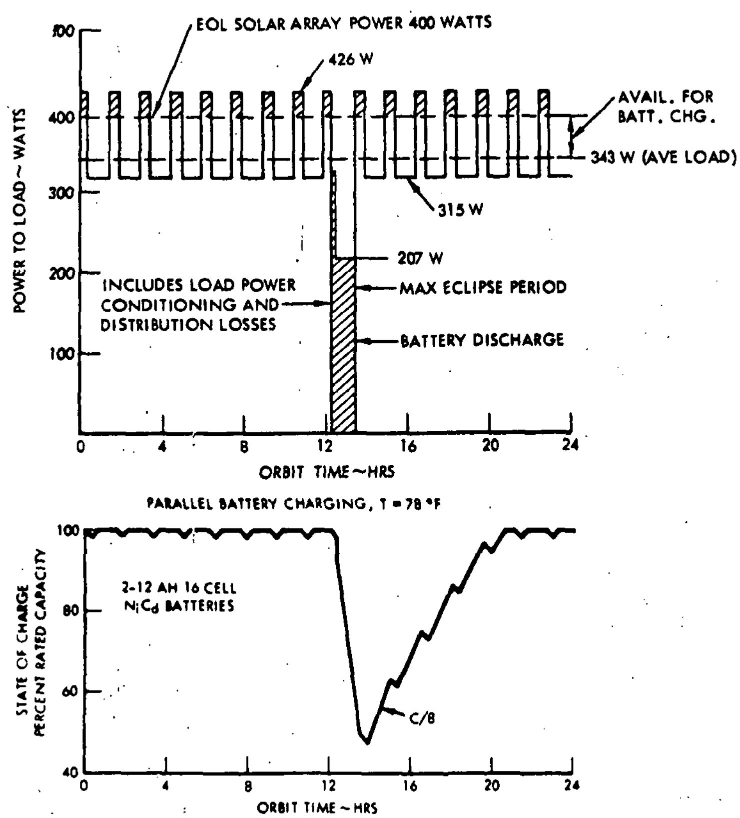


Figure 1-30. Time Required to Charge Batteries

Table 1-17. Electrical Power Subsystem Weights

Components/Assemblies	Weight lb	Weight kg	Potential Supplier
SOLAR ARRAY	(58.6)	(26.6)	
Panels (2)	38.6	17.5	EOS, Ferranti
Drive mechanism (2)	15.0	6.8	BBRC, Spar, G.E.
Linkage and fitting (2)	5.0	2.3	NR
POWER COND. & DISTRIBUTION	(52.7)	(23.8)	
Charge and discharge	11.3	5.1	G.E.
Central control and logic	5.1	2.3	G.E.
Packaging	4.9	2.2	G.E.
Shunt dissipators	2.4	1.1	G.E.
Amp-hour meters	4.0	1.8	Engr. Magnetics
Power conditioning	5.0	2.3	
Cabling	20.0	9.1	NR
ENERGY STORAGE			
Batteries (2)	44.3	20.1	G.E.
Total	155.6	70.5	

1.6 ATTITUDE STABILIZATION AND CONTROL SUBSYSTEM (ASCS)

The decision to use three-axis stabilization as the basic mode of operation allowed a great deal of creativity in the design of the spacecraft which ultimately was translated into the flexible high capacity telecommunications relay system discussed previously. The design of the system and the sizing of its elements were governed primarily by the characteristics of the disturbance torques and system reliability goals. System analysis activities indicated pointing accuracy requirements imposed by the telecommunications function were not severe, since all antenna beams are steerable, and therefore had minimum influence in the design activity. Of greater importance is the need for accurate knowledge of spacecraft attitude to establish a reference for pointing the antennas for S-band.

The baseline ASCS has two functionally independent modes of operation. The first is the transfer orbit mode which uses spin stabilization. The second mode is the operational mode on orbit which uses momentum bias/momentum transfer three-axis stabilization. The momentum bias approach counters the cyclical components of the solar pressure torques and the reaction torques from the gimbaled antennas with the minimum expenditure of propellant.

A block diagram of the complete ASCS is presented in Figure 1-31. A summary of key performance requirements and system parameters is presented in Table 1-18. All system components are existing flight-qualified hardware with the exception of minor changes in the horizon scanners. In addition to being a high performance design approach, the main attribute is high reliability and the absence of any single point failures.

Spin stabilization (90 rpm) is employed from booster separation until after apogee motor firing. Established flight control techniques were selected for this phase. Active nutation control utilizing accelerometers to sense nutation and reaction jets to provide correction torques are used for this minor axis spinner. The spin vector precession maneuver is controlled using sun sensor pulses for proper phasing of reaction jet torque commands. Attitude determination is performed using body-mounted spin scanned horizon sensor and digital sun sensor data. The effect of propellant motion on nutation stability has been examined and is not expected to be a problem.

The baseline 3-axis stabilization system employs two momentum wheels with integral earth horizon scanners in a shallow "V" configuration and a gas bearing gyro/nutation damper normal to the V-plane (see Figure 1-32). This configuration was selected over the other approaches primarily because its freedom from single point failures gives an appreciable reliability advantage over the other candidates. The momentum wheels provide a nominal momentum bias of 12.5 ft-lb-sec (17.1 kg-m^2/sec) along the Y axis. The momentum bias and its natural quarter orbit coupling permit passive yaw control during coasting flight. Pitch control is obtained by driving the wheels in unision. Differential wheel speed control is used to transfer momentum into the Z-axis to obtain active nutation damping and roll control. The two-degree-of-freedom gas bearing gyro provides active yaw control during delta-V maneuvers. The variable speed gyro can also serve as a backup nutation damper by transferring a small amount of momentum into the spacecraft Z-axis. The gyro

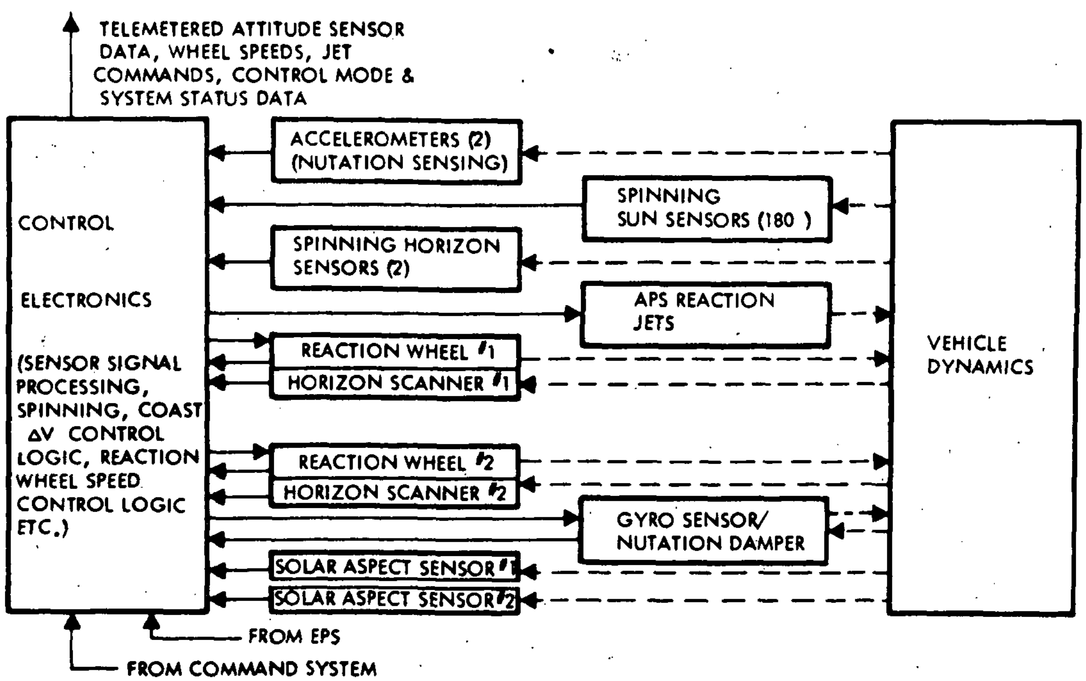


Figure 1-31. Attitude Stabilization and Control System

data also may be used in a backup gyro-compassing mode to aid in attitude determination in the event of failures in the primary attitude determination sensors. This gyro is normally left with power off except when performing delta-V maneuvers or a contingency function.

Table 1-18. ASCS Key Performance Requirements and System Parameter Summary

Attitude Determination Accuracy (Knowledge)	
Roll: 0.25 degree	(.0044 rad)
Pitch: 0.25 degree	(.0044 rad)
Yaw: 0.25 degree	(.0044 rad)
Spacecraft Attitude Pointing Accuracy:	$\pm 0.58^\circ$ (.0101 rad)/axis
ASCS System Weight:	57.7 pounds total (26.2 kg)
Spin control:	6.6 lb (3.0 kg) (0.85% of SOI weight)
3-Axis stabilized:	51.1 lb (23.2 kg) (6.58% of SOI weight)
Propellant Requirements:	55.3 pounds (25.1 kg)
Spinning attitude control:	8.2 pounds (3.7 kg)
3-Axis attitude control:	1.5 pounds (0.7 kg)
Orbit change delta-V maneuvers:	45.6 pounds (20.7 kg)
Stationkeeping Accuracy:	$\pm 0.125^\circ$ (.0022 rad) (corrections approximately every 17 days)
Momentum Storage and Subsystem Requirements	
Momentum bias:	12.5 ft-lb-sec (17.1 kg-m^2/sec)
Maximum cross axis momentum transfer:	± 0.2 ft-lb-sec (.27 kg-m^2/sec)
Momentum Dumping Maneuvers:	No more than once per day

This basic concept is presently being developed at NR for use on a military satellite that will be flown in early 1974. The stabilization and control system design and spacecraft stability have been thoroughly analyzed using digital simulation programs and through use of an air-bearing table on which a similar configuration was tested. Rigid body dynamics were used to determine the effects of antenna slewing on spacecraft attitude and a structural dynamics analysis using modal techniques is presently being performed to verify the validity of assumptions and conclusions of the previous analyses. NR has considerable experience in the design and analysis of flexible structural bodies in the Apollo command and service module docking and Saturn S-II launch vehicle dynamics, as well as on-going unmanned spacecraft designs, that allows for a great deal of confidence in the dynamic analyses conducted in this study.

Three-axis attitude determination is performed at the ground station utilizing horizon sensor and solar aspect sensor data. The selected momentum wheel/infrared earth scanner assemblies use the wheel motion to scan the earth. This approach provides horizon sensing to better than 0.1° (.0017 rad) accuracy at minimum weight and cost. Digital solar aspect sensors mounted on the solar arrays provide the remaining data necessary for three-axis attitude determination to better than 0.25° (.0044 rad) in each axis.

The dominant disturbance torques acting on the TDRS are solar pressure (long term) and antenna motion (short term). Digital simulation programs were used to study these effects and to develop sizing requirements for the momentum storage subsystem. Analysis of the TDRS pointing accuracy requirements indicate that they are less severe than for most contemporary communication satellites. Attitude determination of 0.25° (.0044 rad) per axis and vehicle pointing accuracies of 0.58° (.0101 rad) per axis are found to be adequate. The baseline system has substantial growth potential in that much better sensing and control accuracies are possible without significant changes in the system components.

The reaction jet thruster arrangement is shown in Figure 1-33. The two-quad 16-jet configuration with initial and final thrust levels of 0.27 lb (1.2N) and 0.09 lb (.4N) satisfies all mission requirements. The system can accommodate the worst combination of two jet failures (with an increase in propellant consumption in some cases). The configuration employs more thrusters than the minimum number possible to prevent physical interference with other spacecraft components.

1.7 AUXILIARY PROPULSION SUBSYSTEM (APS)

The auxiliary propulsion subsystem includes the spacecraft reaction control subsystem and the apogee injection motor. In both cases, state-of-the-art technology was used and in all instances flight-proven equipment was selected, needing no flight qualification testing.

1.7.1 Reaction Control Subsystem

Figure 1-34 is a schematic of the RCS key features which are:

1. It operates in a blowdown mode (3.35 to 1).
2. Separate gaseous nitrogen fill/drain valves avoid loss of total pressurant if a leak occurs in one tank.
3. Either propellant tank can be isolated should a leak occur.
4. Each thruster is equipped with a redundant (two seats in series and two coils in parallel) propellant flow valve.
5. Any individual thruster can be isolated in the event of a "runaway" thruster if the redundant valve fails open.
6. The entire subsystem is made up of existing and proven components.
7. The 16 thrusters are housed in two identical and compact modules. All maneuvers can be accomplished after any two thrusters fail.

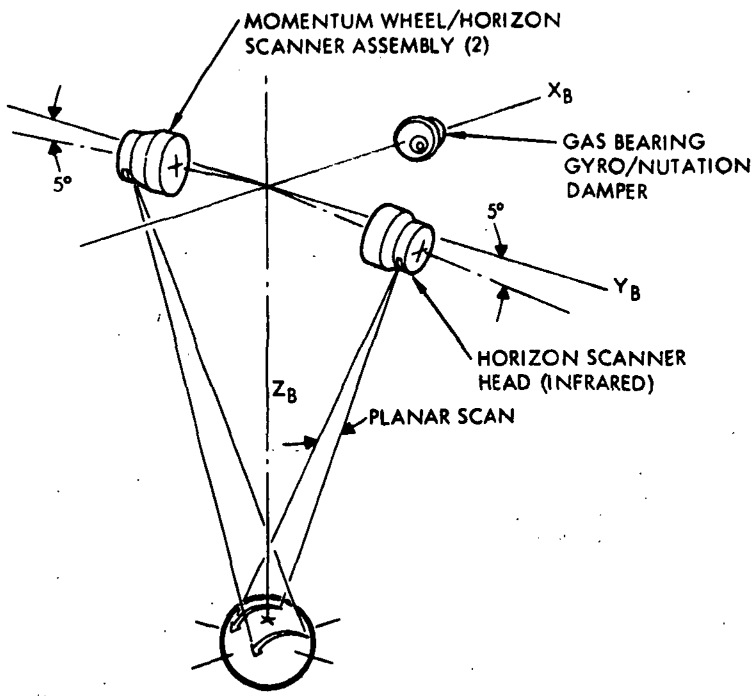
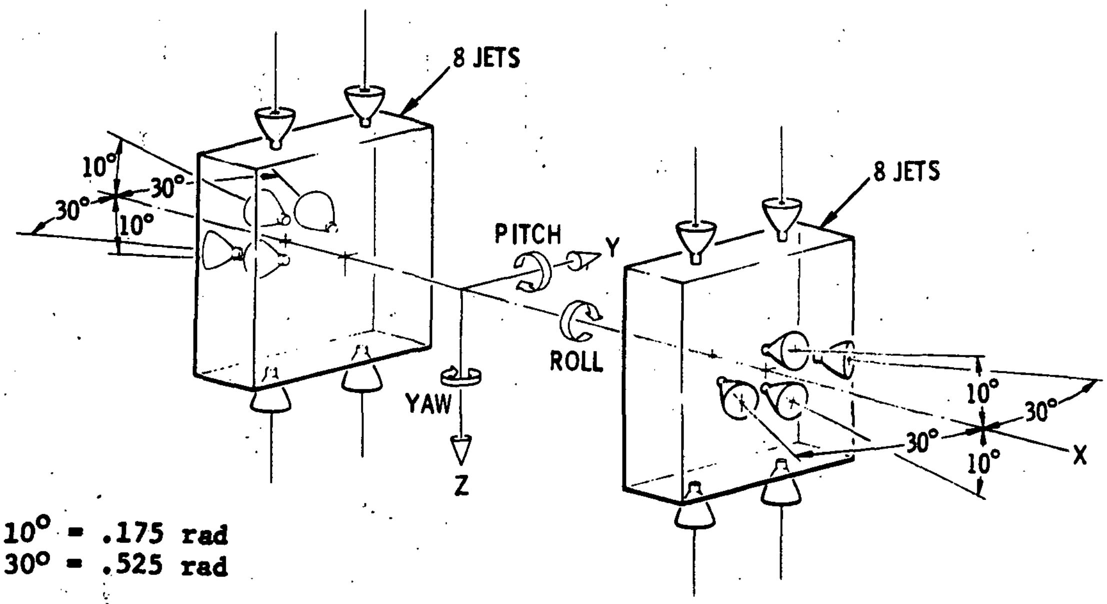


Figure 1-32. Momentum Storage Subsystem Arrangement



10° = .175 rad
30° = .525 rad

Figure 1-33. APS Engine Arrangement

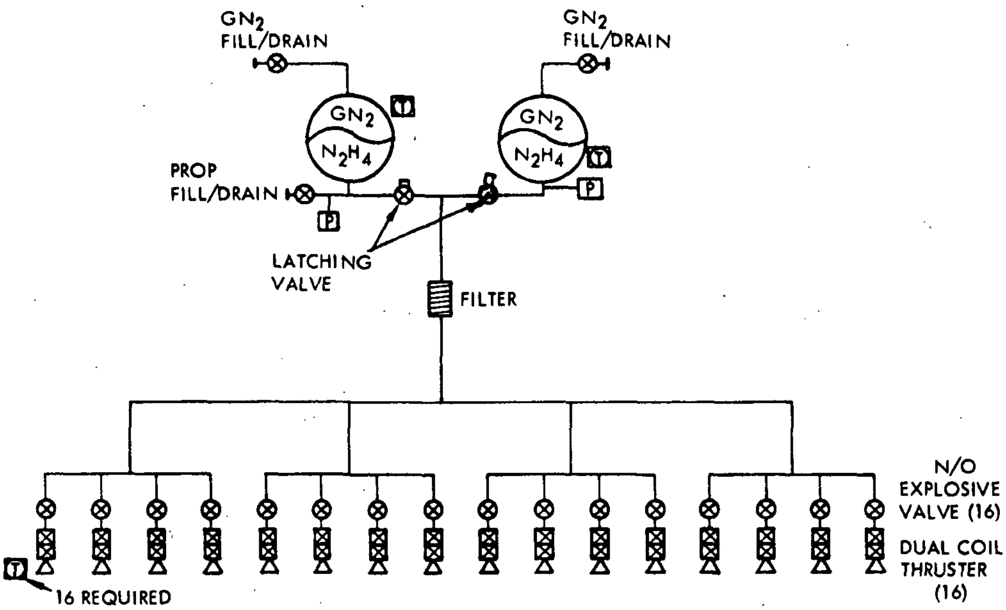


Figure 1-34. Auxiliary Propulsion System

To satisfy the five-year operational requirement an improved elastomer composite, ethylene propylene terpolymer (EPT-10) was chosen as the expulsion diaphragm material. This material is now in its third year of long-life testing by Pressure Systems, Inc. All evidence indicates that a five-year-life EPT-10 diaphragm will be demonstrated before the TDRS enters the implementation phase. Current programs using or planning to use EPT-10 include Pioneer F&G, ATS F&G, MVM, CTS, AERO-ERNO, P-95 (military), Viking Lander and the 4.5-year Jupiter/Saturn mission.

Further, in the unlikely event that the long-life characteristics of elastomer diaphragms are not demonstrated, a back-off design approach was synthesized that meets all mission requirements, but with a 12 lb (5.4 kg) weight penalty. The back-off position uses the hydrazine/catalytic thruster design presented here for all short-term functions through longitudinal station acquisition. An ammonia/resistojet system would also be installed that can handle all long-term attitude control, stationkeeping, and station-change requirements. However, it must be emphasized that this eventuality is not expected to occur.

Another long-life consideration is the thruster catalyst. A large number of "cold" starts could impose serious damage to the catalyst as a result of the startup pressure overshoot. The solution to this problem incorporates active thermal control directly on the catalyst bed. Data from thruster manufacturers indicates adequate protection is provided if the bed is heated to 300F (149C) prior to each start. A power requirement of 0.7 watt for two to three hours will heat the bed to 300 F(149 C).

Verification of the APS performance and determination of propellant quantity will be accomplished through measurements of tank pressure and temperature. This is a standard, proven method. Unused propellant will be determined by calculation using the PVT relationship.

The weight of the propulsion system hardware and trapped propellant is 38.4 lb (17.4 kg). Useful propellant weight is 55.3 lb (25.1 kg). Table 1-19 shows the RCS propulsion requirements established by the stabilization and control analysis and the resulting propellant requirements.

Table 1-19. Propellant Requirements

	Impulse lb-sec (N-sec)	Propellant lb	(kg)
Precess to SOI attitude (150 degrees)	1212 (5400)	5.5	(2.5)
Nutation control-transfer orbit (30 hours)	121 (540)	0.7	(0.3)
Despin (from 90 rpm)	442 (1960)	2.0	(0.9)
Acquire local vertical (5 times)	188 (835)	0.9	(0.4)
Correct apogee injection error (3σ)	4350 (19300)	19.8	(9.0)
Stop drift at final station (5 degrees/day)	1100 (4890)	5.0	(2.3)
Longitudinal stationkeeping (5 years)	818 (3630)	3.7	(1.7)
Momentum dumping (5 years)	117 (520)	0.6	(0.3)
Longitudinal station change 2 to 65 degrees - 15 days	3770 (16750)	17.0	(7.7)
Total		55.3	(25.1)

1.7.2 Apogee Motor

A modified version of the Thiokol TE-M-616 solid motor was selected as the apogee injection motor. It is presently in development for the Canadian Communication Technology Satellite (CTS) and first flight article delivery is expected in November, 1972. Modifications of the motor include a 6-in. (15.2cm) reduction in nozzle length which reduces inert weight 7 lb(3.2 kg) and off loading 46 lb (20.8 kg) of propellant. Both changes are minor and no requalification is required. A summary of the modified CTS apogee motor characteristics is given in Figure 1-35, and a summary of its performance in conjunction with the Delta 2914 was presented previously.

1.8 THERMAL CONTROL

The thermal control system accommodates induced heat loading from the apogee burn and motor case heat soakback and from the broad range of equipment power dissipation. The system is sized for a combination of operational and standby housekeeping modes, and the seasonal changes of the long-duration mission. The design employs standard techniques similar to those used on other satellites, and considers seasonal solar load variations on louvered radiator panels. During transfer orbit and eclipse standby operation when power dissipation is minimal, makeup heating maintains the RCS at operational temperatures. The system performs during all mission phases without constraining attitudes, maneuvers or durations.

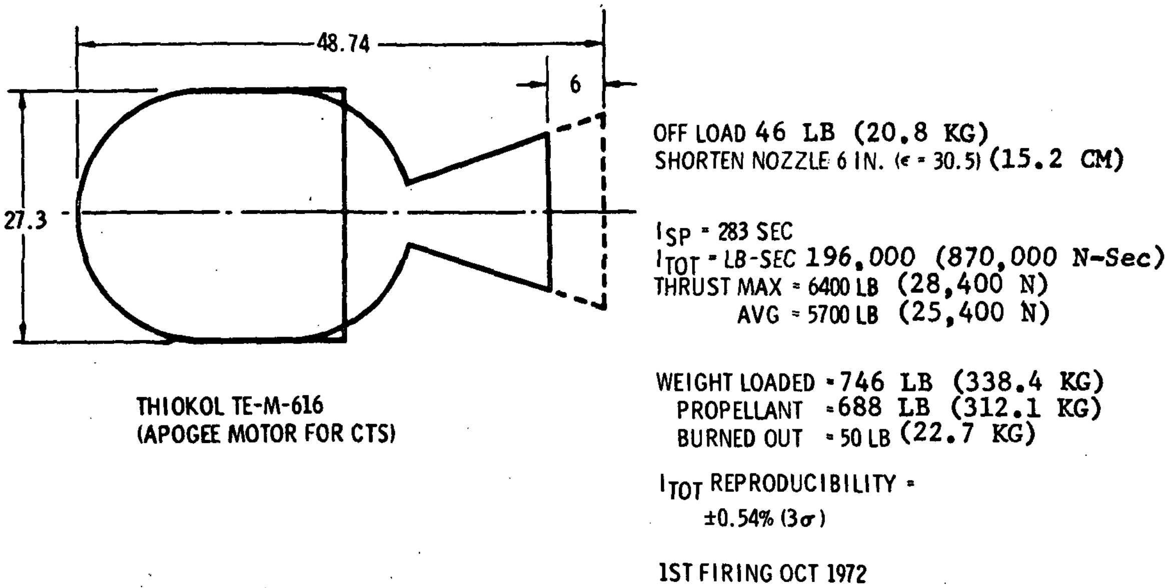


Figure 1-35. Apogee Motor

The mission thermal requirements are summarized for each mission phase in Table 1-20. The TDRS upper temperature is derived from the communications equipment needed to provide high reliability and efficiency without weight penalty. Therefore, the operational temperature limit of the communications subsystem electronics was established at 40 C (104 F). The TDRS lower design temperature limit of 40F (4.4C) is derived from the freezing point of hydrazine. The limit applies to all RCS components to provide continuous operational capability. The battery temperature limits are established from stringent requirements to limit energy storage degradation.

The thermal control design is affected by the following problem areas:

1. The TDRS toroidal body offers limited surfaces for efficient heat rejection due to the daily solar incidence.

2. The TDRS equipment is separated from the heat rejection surfaces.

3. The transfer orbit and the spare TDRS operating modes are powered down.

Table 1-20. Mission Thermal Requirements

Phase	Duration	Environment	Considerations	Requirements
Prelaunch checkout	Hours, as required	Hot or cold day Full power load	Air flow provisions	Provide duct ports
Launch and ascent	4 minutes	Delta fairing heat soak-back, depressurization	Fairing rises to +400 F (204 C)	Jettison fairing, provide vent
Parking orbit	20 minutes	100 n mi (185 km) earth orbit; Min power loads	Fixed attitude, non-uniform heating Stored panels and antennas	
Transfer orbit	24 hours	Solar, with earth emission and albedo near perigee Minimum power loads	90 rpm spin-up, thrust axis attitude orientation maneuver for SOI; stowed configuration	Limit cooldown of TDRS and apogee motor
Insertion and pre-operations	1 to 2 hr	Solar, power-up	Deployment of panels and antennas	Apogee motor case heat soakback into spacecraft
Synchronous orbit	5 years	Solar, operational nominal and reduced power loading	Fixed earth orientation. Once a day roll to sun. 26° (.45 rad) seasonal out of orbit plane sunline change. Up to 72-minute eclipse.	Maintain subsystems design temperature. Reject heat loads. Limit eclipse cooldown

The TDRS body thermal control design consists of (1) 14.4 ft^2 (1.3 m^2) of louvered panel area divided equally between the four equipment quads and located on the north and south surface, (2) multi-ply insulation, and (3) solar reflector thermal control coatings on the louver base and on the insulation blanket. The performance of the system is shown in Figure 1-36 for limiting insulation performance values. The design performs within the control range for normal operation, but the spare TDRS power load is insufficient to balance the heat leakage at the required limit temperature. Cooldown also occurs in transfer orbit for a combination of reduced power load and solar heating. Without constraining the TDRS sunline angle or powering up beyond the demand load, temperatures below allowable occur. Makeup heating is required for the RCS. The additional RCS power requirement is available because of the reduced TDRS demand during these two phases; makeup amounts to 18 watts and 73 watts for transfer orbit and for the spare TDRS in eclipse, respectively.

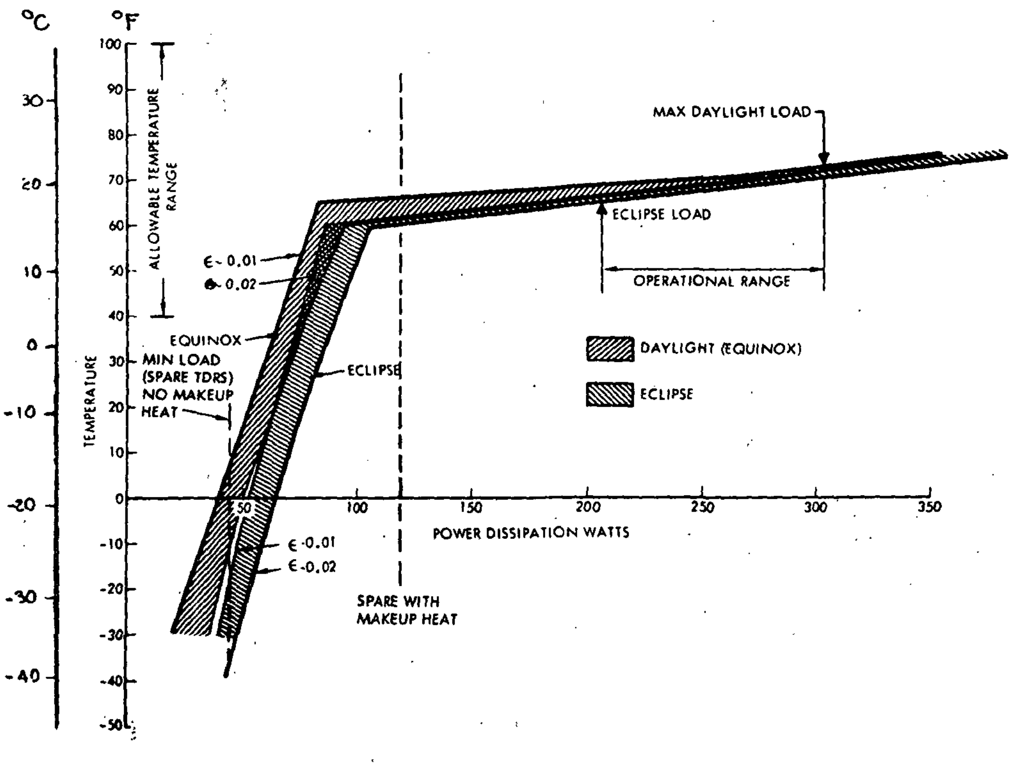


Figure 1-36. TDRS Mean Temperature

Exposed structure, arrays, antennas, etc., are passively controlled within design limits and gradients. Insulation is applied to masts and feeds to maintain alignment. The RCS thruster catalyst chambers are heated to 300 F (149 C) before operation for easy starts. Heat soakback from the expended apogee motor case is attenuated by high temperature insulation lining the tunnel.

A thermal math model was developed to aid in establishing equipment locations. The results of the simulations provide thermal design criteria and preliminary values of equipment temperature.

1.9 RELIABILITY

The major reliability goal set for the TDRS design effort was to provide a satellite reliability of 0.8. This goal was established in conjunction with the GSFC Project Office after preliminary system reliability analyses showed the relationship between satellite reliability and the probability of mission success, which was defined as having one or two satellites remaining at the end of five years. This relationship is shown in Figure 1-37. In developing the curves a booster reliability of 0.95 and an apogee motor reliability of 0.98 were used.

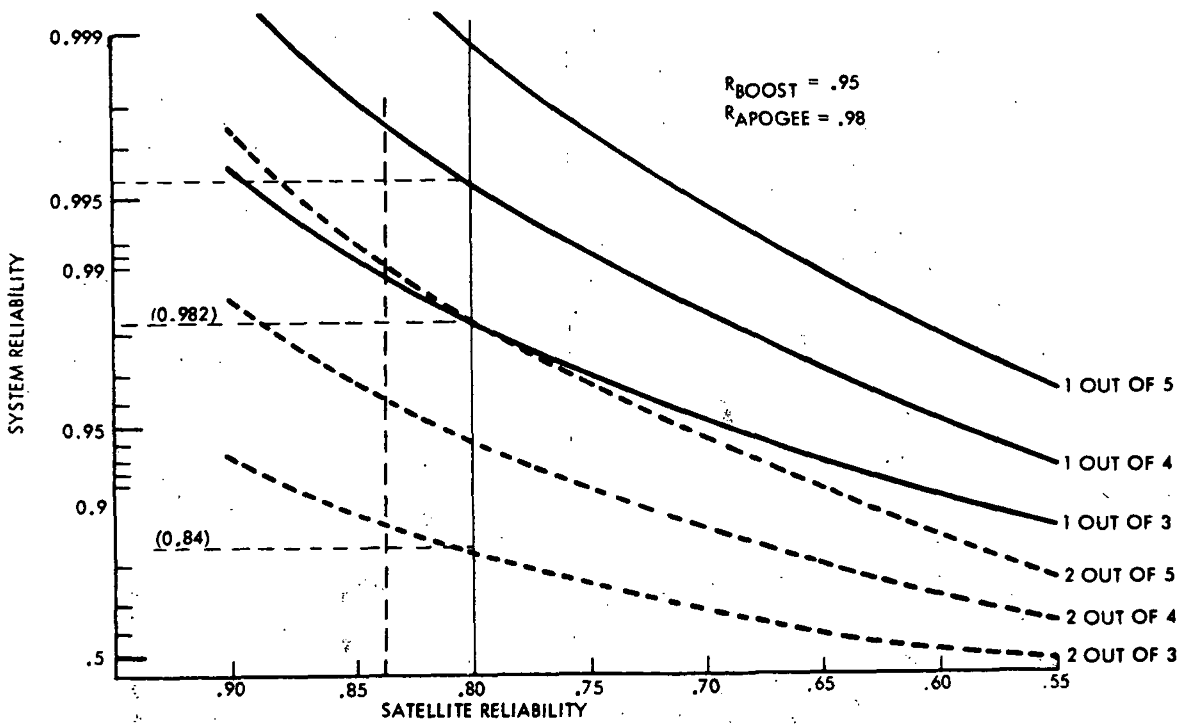


Figure 1-37. System Reliability vs. Satellite Reliability

The curves show the probability of mission success where mission success is defined as the ability of each TDRS to service 20 LDR users and 2 MDR users on the return link and 2 LDR and 2 MDR users simultaneously on the forward link. Reduced forward link capability is permitted during eclipse. This capability exceeds that required in the statement of work and a higher probability of success is obtained using the SOW capability requirement of one MDR user on forward and return and one LDR user on forward link. An even higher reliability will occur for reduced operations below the SOW capability since the satellite in nearly all cases degrades gracefully, allowing mission continuation.

Subsystem reliability analyses show that for the excess capability case, the satellite reliability is 0.804 and for the SOW capability the satellite reliability is 0.844.

Table 1-21 shows the system reliabilities taken from Figure 1-37 of one or two spacecraft remaining in <u>full</u> operation at the end of five years for these satellite reliabilities.

Table 1-21. Probability of Mission Success

Satellite Capability	Probability of Success			
	Excess Capability		SOW Capability	
No. S/C in Full Operation	1	2	1	2
3	.983	.840	.990	.880
4	.996	.950	.998	.965
5	.999	.983	.9995	.991

These high goals were achieved by adhering to a design philosophy throughout the spacecraft of minimizing single point failures, using high reliability components, and using redundancy whenever necessary. The weight margins provided by the selected design approaches permitted the use of redundancy in all critical areas.

A reliability predictive analyses defined potential problem areas and indicated where redundancy or a modified design approach was worthwhile. Logic diagrams were prepared for each subsystem and the reliability values obtained are shown in Table 1-22.

Table 1-22. Reliability Prediction

Subsystem	Predicted Reliability
TT&C	.966
Communication	.915
Structures and mechanical	.999
Attitude	.962
Auxiliary propulsion	.997
Electrical power	.962
Thermal control	.999
Total satellite	.804

The failure modes and effects analyses and the resultant design changes eliminated all critical path failures that would seriously impact spacecraft reliability. In those few instances where single point failures were retained, the probability of occurrence was minimal and in addition the cost, weight and complexity of elimination made their removal impractical.

1.10 USER TRANSPONDER DESIGN

1.10.1 LDR Transponder

A simplified schematic block diagram of the LDR transponder is shown in Figure 1-38. The receiver is designed to tune to any one of four UHF carrier frequencies used by the two TDRS's. The carrier in the forward link is phase modulated by a 167 K chip/sec pseudo noise (PN) sequence to discriminate against multipath and to distribute the signal energy radiated from the TDRS to conform to the IRAC requirements. A 1 Mchip/sec PN sequence is used in the return link to permit code division multiple access of 20 LDR users through a common channel in the TDRS with sufficient process gain to achieve a specific level of performance for any one user in the face of 19 interfering users. The use of PN sequences in both the forward and return links provides a vehicle for deriving range information.

The data is delta PSK modulated with convolutional encoding to save power through error control. The demodulation process in the receiver is rather straightforward. The transmitter, except for the PN modulation, is a standard transmitter configuration.

A preliminary estimate of the size and prime power requirements of the LDR transmitter and receiver are: transmitter - power, 16 watts dc power; size, 225 in^3 (3687 cc); and receiver - power, 12 watts; size, 195 in^3 (3195 cc).

1.10.2 MDR Transponder

The receiver is a single channel S-band receiver. The carrier in the forward link is modulated by a single 5 Mchip/sec PN sequence during the code acquisition phase to distribute the signal energy radiated from the TDRS to conform to IRAC requirements. During the ranging phase a second PN sequence of 500 Kchip/sec rate is modulo-2 added to the first. A 500-Kchip/sec PN code generator synchronized to the uplink 500-Kchip/sec code is used to modulate the downlink carrier.

In both manned and unmanned users the data is delta PSK with convolutional encoding for error control. In the manned user case, the data and delta modulated voice are time division multiplexed to form a serial data stream. A simplified schematic diagram of the MDR transponder is shown in Figure 1-39.

A preliminary estimate of the size and prime power requirements for the MDR transmitter and receiver are: transmitter - power, 33 watts dc power; size, 240 in^3 (3933 cc), and receiver - power, 12 watts; size, 205 in^3 (3359 cc).

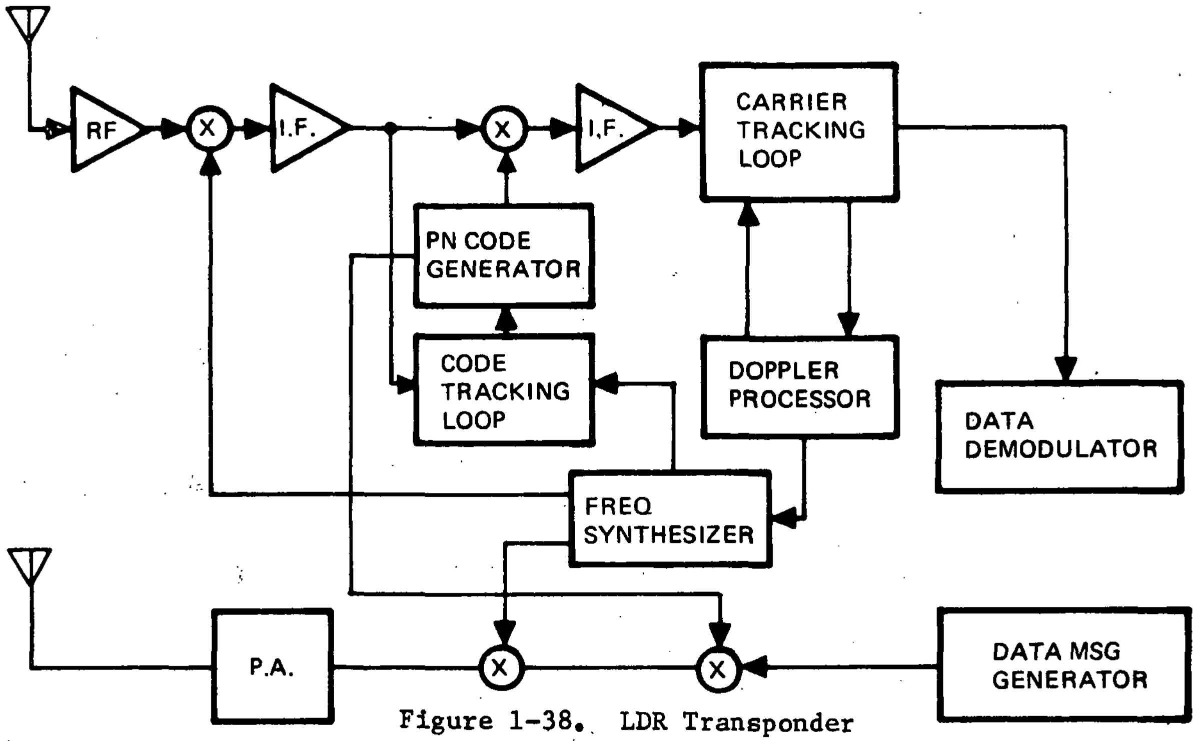


Figure 1-38. LDR Transponder

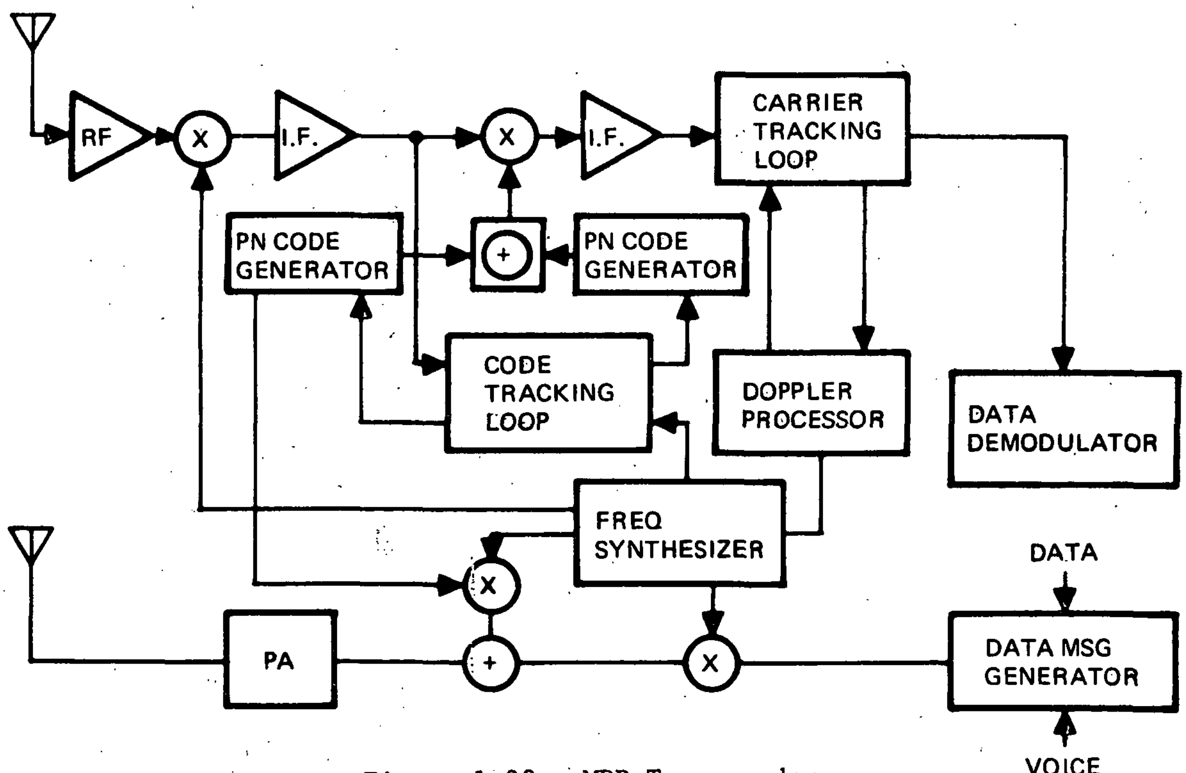


Figure 1-39. MDR Transponder

1.11 NETWORK OPERATIONS AND CONTROL

Primary emphasis in the network operations and control studies was to achieve a real-time operational system. At the same time, practical considerations required maximum use of existing and planned facilities and organizations for cost-effectiveness. The operational studies consisted of:

1. Development and description of a TDRS system concept

2. Description of the operational and functional interfaces between primary TDRS system elements for network operations

3. Functional analysis of TDRSS operations and development of functional flows for all important functions and operations to a third/fourth level of detail

4. Development and description of a sequence of events and operations performed by the TDRS system elements in a representative operational phase mission of two operational TDRS spacecraft providing service to LDR and MDR user spacecraft.

Figure 1-40 shows the TDRS system concept which operates and controls the TDRSS network in real time. The figure shows three groups of elements: GSFC-located ground elements, remote-located ground elements, and space elements. The space group consists of the two operational TDRS spacecraft at geosynchronous altitude with two-way communications links to the RF ground station, and the low data rate and medium data rate user spacecraft at low earth orbit, including the space shuttle.

The ground system provides and maintains real-time communications between all elements. It also implies reducing long land lines as much as possible since these are both costly and complex. Larger users who now control their own spacecraft will continue to do so with TDRS and can now be given real-time capability. Small users who now operate through the system elements for spacecraft control and who do not require real-time control may continue either with the other elements or with TDRS to maintain routine operations but with provisions for TDRS to take care of urgent requirements.

The ground system will have GSFC-located and remote-located elements. The TDRS RF ground station very likely will be remote-located because of the large (three 60', 18.3 m) antennas and facilities involved. It is shown to be at Rosman, N.C., although it can be at any other desired location. Other remote elements are MSC at Houston, some user control centers, Launch Control Center at KSC, and STDN tracking stations. All these remote elements are linked directly to the Network Operations Control Facility at GSFC by NASCOM communication lines.

The Network Operations Control Facility at GSFC is a centralized operating organization with hard line communications links connecting the internal elements. It capitalizes upon the existing location at GSFC of many user control centers and of substantial facilities already in the GSFC complex suitable for TDRSS operation. GSFC elements include: (1) a Network Ops Control Center to

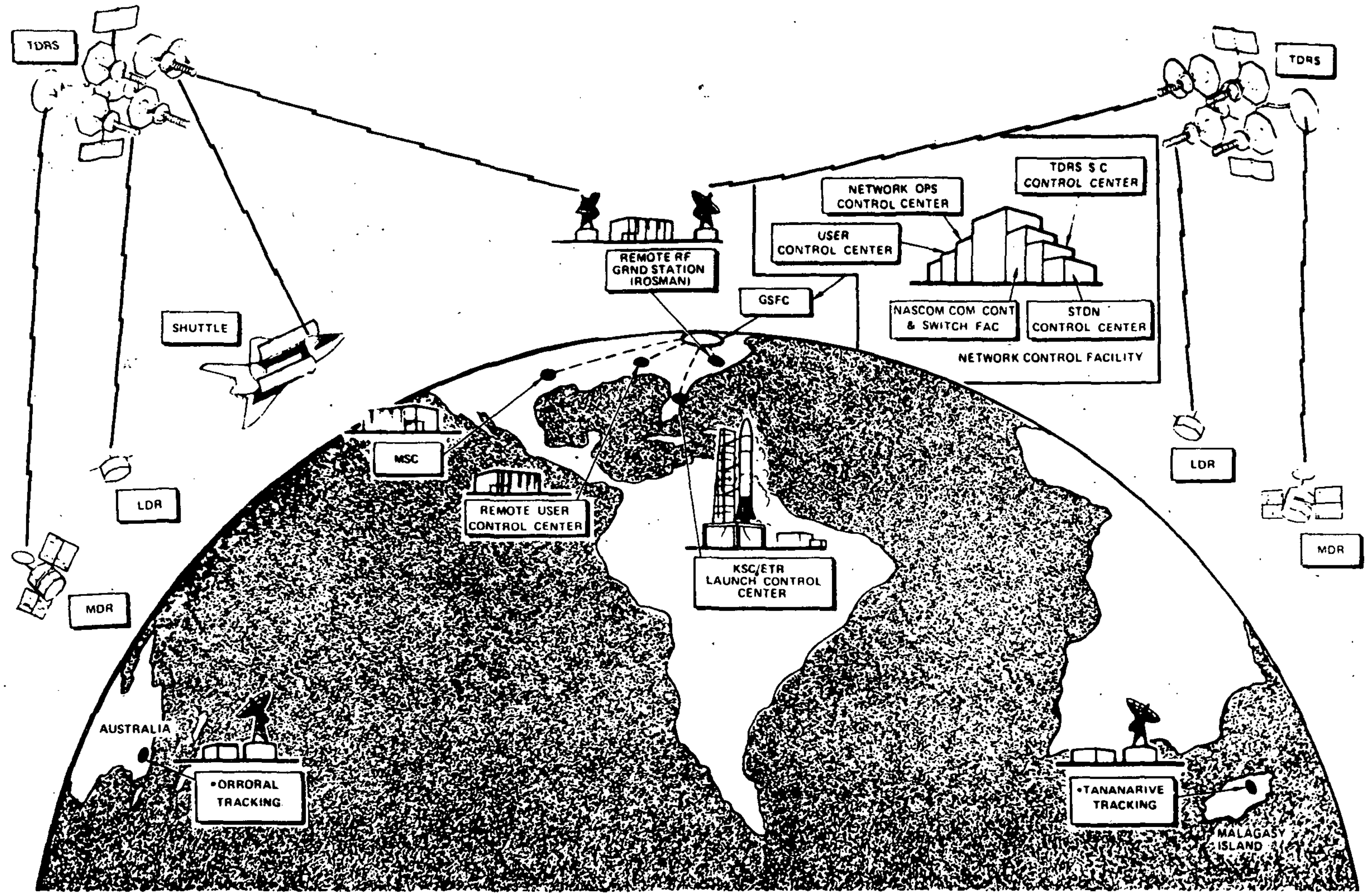


Figure 1-40. TDRSS Operational Concept

interconnect and control the network and schedule its operations; (2) a TDRS Spacecraft Control Center to control the TDRS; (3) User Control Centers to control individual user spacecraft; (4) a NASCOM Communications Control and Switching Facility to handle, switch, and direct the flow of communications to and from the various internal elements of the GSFC facility and to and from the remote elements; and (5) an STDN Control Center to direct and monitor the STDN support functions.

The large users at GSFC will have an operations control unit to manage their own service requirements, allowing them to proceed directly through a communications processor for command multiplexing, priority routing, data encoding, and high-speed modem operation to the ground station. Scheduling will be rapidly developed and provided to effect real-time implementation. Routine operations can be maintained by small users and others by using mail, TTY, and telephone facilities and the implementation by other operating centers.

Figure 1-41 illustrates a typical TDRSS network with ground station remotely located at Rosman, and a central network control facility at GSFC. The primary system elements and the operational and functional interfaces between them are shown. Automated and hard line interface links exist between all elements of the control ground network facility to facilitate fast response time and real-time operations. Ground communications are provided by the NASCOM network which furnishes all NASA mission control, technical control, and computation centers with access to remote tracking, data acquisition, and command stations. A highly automated ground facility is used including computer control and processing as well as associated software, although the degree of centralization and automation of the data handling system has not yet been determined.

Ground link interface definition depends significantly on the diversity of the ground elements. The degree of ground station separation depends largely on the data handling requirements of broadband users, the need for data processing facilities, and the costs and real-time limitations of long line transfer and processing of data. Commands will be generated at the user control centers and forwarded through NASCOM to the TDRS ground station and appropriately formatted for transmission to the spacecraft. The primary elements involved in the operational process are shown in the figure with the flows between them and the principal functions of each element.

The NASCOM Communications Control and Switching Facility manages, controls, switches, codes and distributes all communications between elements in a format appropriate to the destination element. Service is provided by TTY, voice, high-speed data, and wideband data. Information may be carried through the transmission system in its original form at baseband, or signal converters may change it to a form more suitable for transmission. Modems (Modulators/Demodulators) may be required at each NASCOM terminal. Messages are routed by identifying addresses at the beginning of each message. The system is computer controlled, employing NASCOM digital computers at the switching centers for routing.

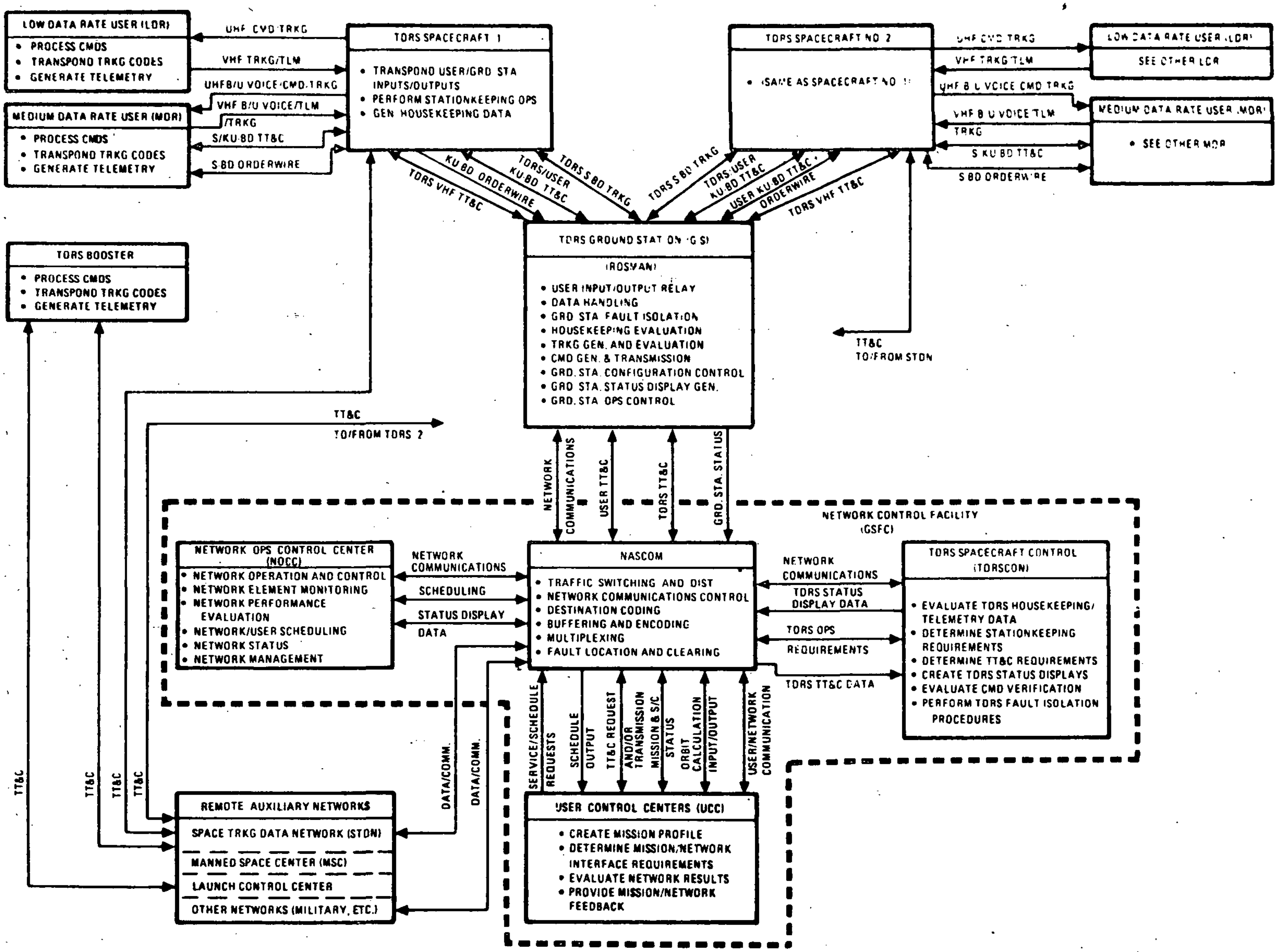


Figure 1-41. Primary System Elements and Their Operational and Functional Interfaces

Interfaces exist between all the network elements via NASCOM. The Network Ops Control Center (NOCC) manages and coordinates all network activities. Its interfaces via NASCOM are for network communications and network operations control, scheduling, and status display. The TDRS Spacecraft Control Center (TDRSCON) manages and controls the TDRS spacecraft and relays communications service to user spacecraft. Its interfaces via NASCOM are for network communications, TDRS status and display, TDRS operations, and TDRS telemetry, tracking and command.

The User Control Center (UCC) manages and controls its user spacecraft and receives and evaluates its mission results. Its interfaces via NASCOM are for network communications, user spacecraft status and display, user spacecraft mission operations and control, and user spacecraft telemetry, tracking, and command. These require interface relationships for the flow of service and schedule requests, status information, tracking and orbit data, and real-time communications.

The TDRS ground station, and its station control, manages, controls and operates an RF system to generate, transmit, receive, and handle RF commands and data to and from space and to and from the ground elements. It interfaces with the ground network elements through NASCOM for network communications, user and TDRS telemetry, tracking and command, and for station control. In turn, the TDRS ground station interfaces between the user and TDRS Control Centers and the user and TDRS spacecraft for the RF operations required for the transmission of commands and the reception of telemetry data. Its functions include: command and tracking generation and transmission; encoding, decoding, and high-speed modem operations; command data buffering and routing; command status and command verification multiplexing; data handling, processing, and conditioning; and RF antenna operations and control; station control and status display; station monitoring and checkout.

The interfaces between the User Control Center and the remote TDRS Ground Station were based on having user spacecraft outputs signal conditioned at the ground station before transmission to GSFC. A message switching ground link interface configuration was used in which several data management functions as well as signal conditioning are performed at the ground station to assure time correlation and quality of all user data and to reduce data rate limitations, time delays, telemetry scheduling requirements, and equipment duplications. A computer at GSFC automatically distributes data messages according to destination codes.

The TDRS interfaces directly with the ground station and user spacecraft. It performs the relay operations to user spacecraft operations to the user via the ground station and NASCOM. The interfaces with the ground station provide two-way communications of TDRS S-band tracking, user Ku-band TT&C and orderwire, and TDRS VHF TT&C information. Interface links with the LDR user spacecraft are a UHF forward link and a VHF return link of tracking, telemetry, and command data. The interface links with the MDR spacecraft provide a two-way S-band or Ku-band link for transmission and receiving of TT&C information, a two-way S-band orderwire, and a backup UHF forward and VHF return link for voice, telemetry, tracking, and command.

Additional interfaces exist between the remote ground elements and the space and GSFC-located elements for tracking, telemetry, and command. The interfaces between these remote elements and NASCOM are essentially operational and functional components of the overall system.

With the system concept and element interfaces as described, a detailed functional analysis of TDRSS functions and operations was conducted to apply the real-time system philosophy to implement the functional objectives. Detailed functional flow diagrams were developed to a third/fourth level of operations. First level functional flows were developed for both preoperational (launch and deployment) and operational (relay mission) phases. The top level and first level functional flow diagrams are shown in Figures 1-42, 1-43, and 1-44. After achieving the proper station, the TDRS performs as a relay to service user spacecraft with two-way communications to the TDRS ground station and respective users. A concurrent responsibility is to assure the cooperative performance and condition of the TDRS spacecraft to allow effective performance of this relay function throughout its lifetime. A third concurrent responsibility is to assure the cooperative, controlling, monitoring, and handling performance and condition of the ground elements. The combined objective is to achieve continuing, effective, real-time operations of the system over the desired five-year lifetime. Hence, in Figure 1-44, the operational mission involves, in a sequential and/or parallel manner, all the desired functions of a TDRS system, namely, command and control, data transfer, tracking, housekeeping, acquisition, handover, stationkeeping, station transfer, and spare spacecraft operations. Detailed functional flow diagrams were developed for each of these functions at second, second/third, and third/fourth levels. These show operational sequences, including alternative sequences, concurrent as well as sequential, for both real-time and nonreal-time operations.

The functional procedures that permit real-time operation are, for example, (1) commands generated in the user and TDRS control centers; (2) commands transmitted directly to a communications processor (at GSFC) for command multiplexing and priority routing; (3) high-speed modem and decoding performed; (4) transmitted via NASCOM to the RF ground station (e.g., at Rosman); (5) high-speed modem and decoding at the ground station; and (6) ground station buffering and routing for transmission to the TDRS spacecraft. Similar operations are performed in reverse for return data, housekeeping, tracking, verifications, etc. The interfaces and functional relationships are compatible with these desired real-time functional procedures.

Figure 1-45 presents a flow diagram of a representative TDRS sequence of operational phase mission operations after TDRS is on station and is contacting the users for the first time. This flow is only representative of the many operational sequences possible with this system. Generally, it assumes a user must be acquired, commanded, tracked, and monitored to assure it is operationally ready to participate with the TDRSS in transferring data. After handover takes place to a second TDRS the data transfer is resumed. Stationkeeping occurs approximately every 17 days. The functions associated with LDR and MDR user acquisition and handover operations are illustrated in Figures 1-46 through 1-49.

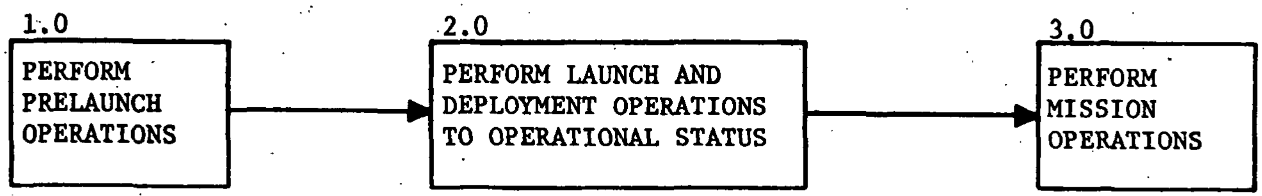


Figure 1-42. Top Level Functional Flow Diagram

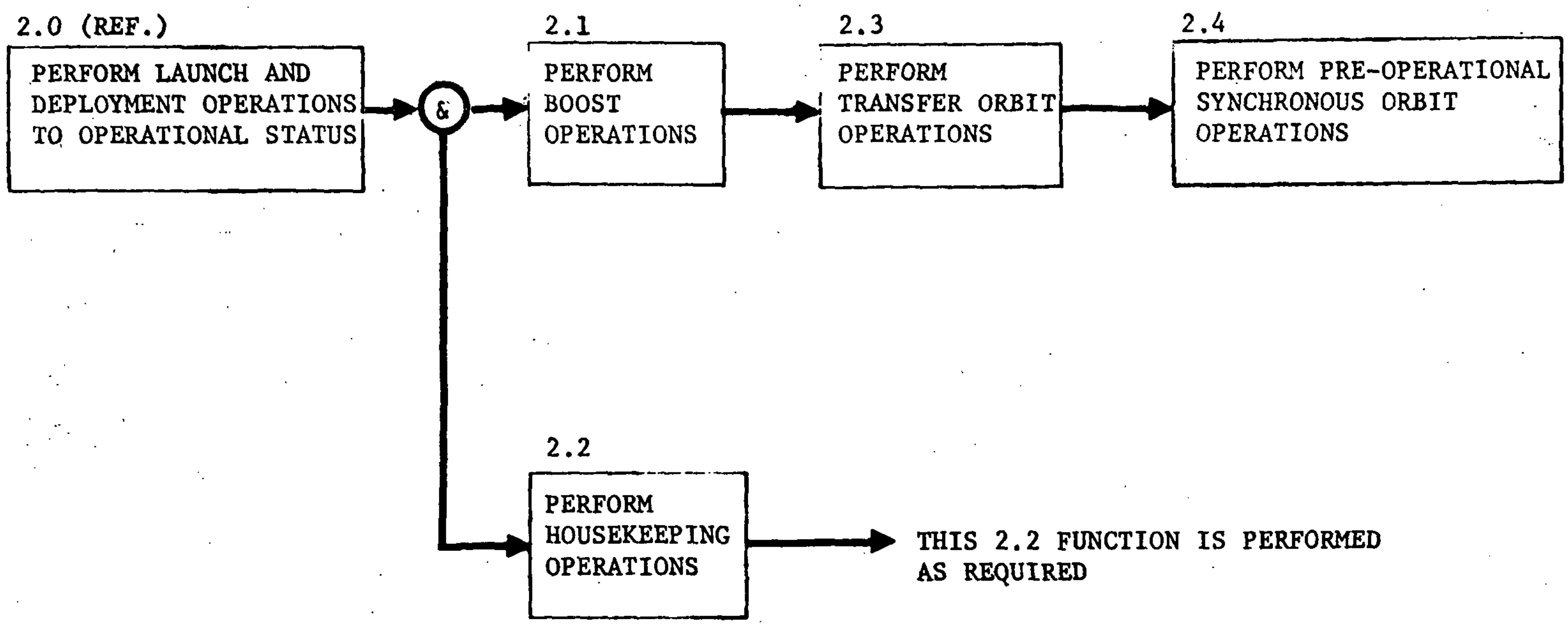


Figure 1-43. FIRST LEVEL FUNCTIONAL FLOW DIAGRAM (2.0 PERFORM LAUNCH AND DEPLOYMENT OPERATIONS TO OPERATIONAL STATUS)

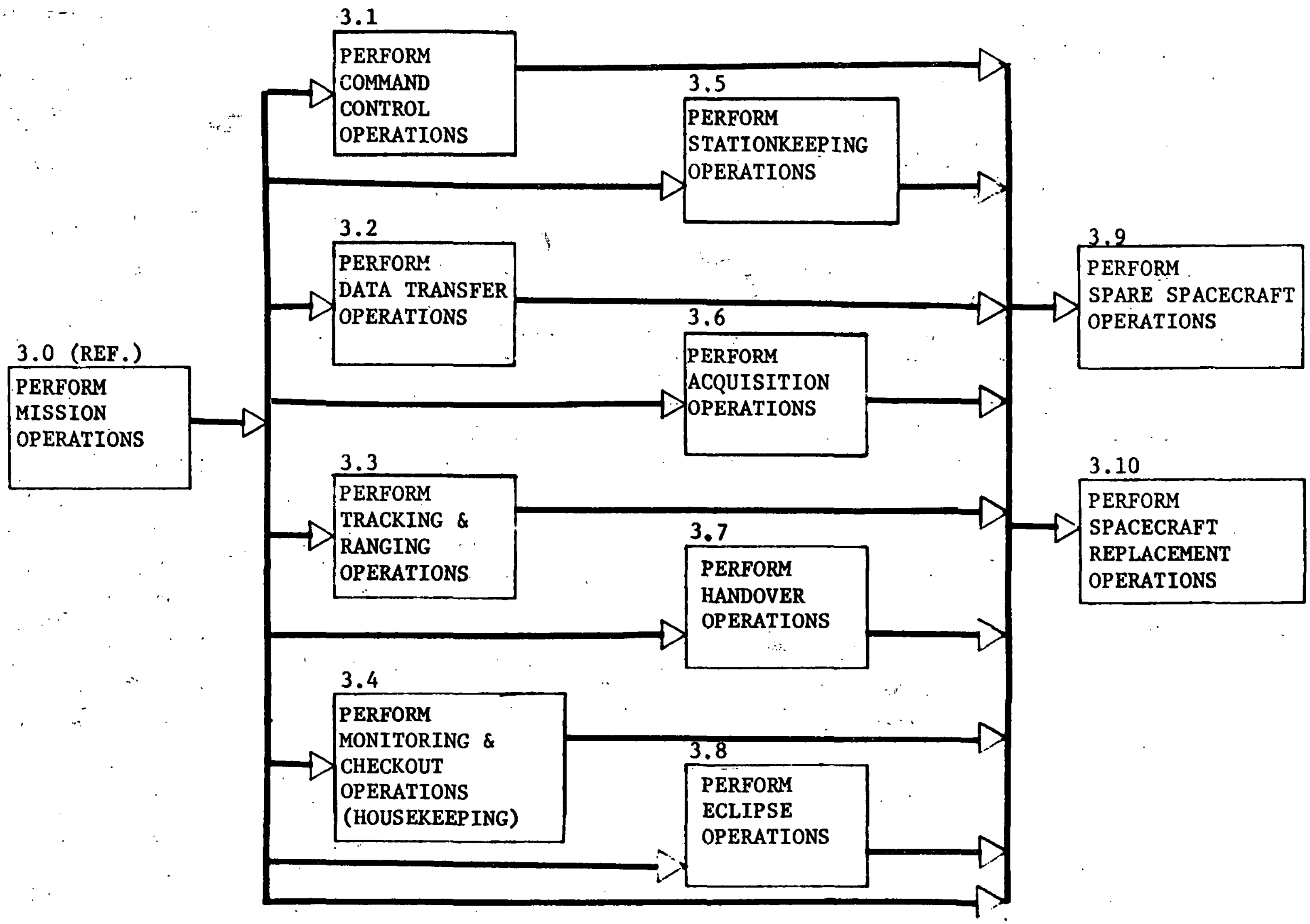


Figure 1-44. FIRST LEVEL FUNCTIONAL FLOW DIAGRAM
(3.0 PERFORM MISSION OPERATIONS)

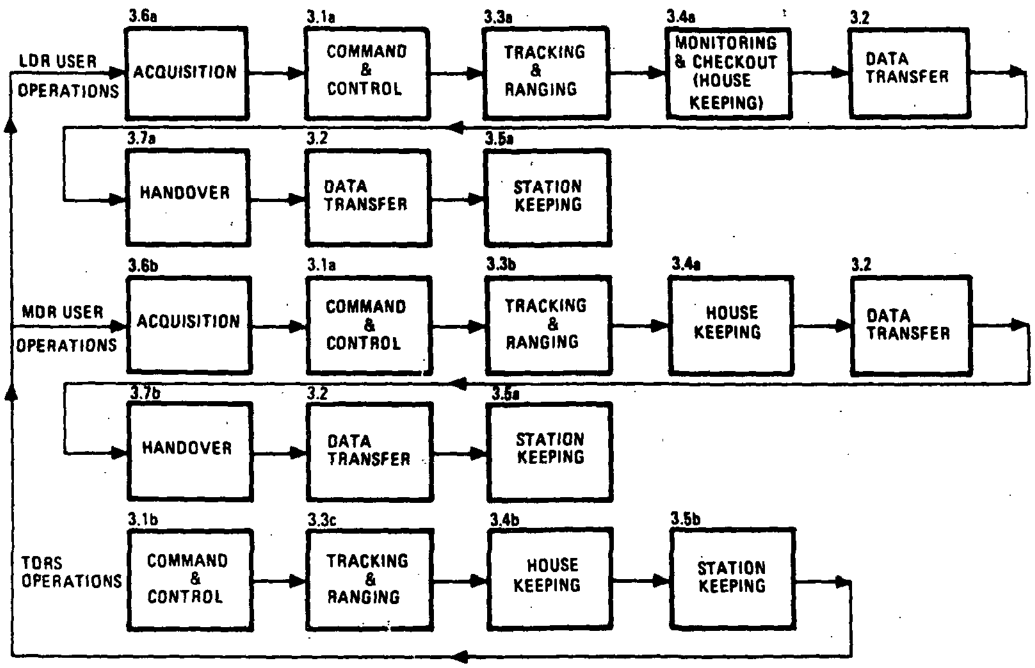


Figure 1-45. Operational Phase - Representative Mission Operations

The procedures defined by the functional flow provide a detailed set of operational procedures for implementing a real-time TDRS system concept, while allowing routine procedures to be carried out where time is not a sensitive factor. They also exert minimum impact on existing or planned organizations and facilities, particularly in the early years of the system operation. Nevertheless, there is inherent flexibility in the system concept and operation to permit growth and modifications to allow for varying degrees of automation, centralization, and sophistication.

1.12 TDRS GROUND STATION

The TDRS Ground Station is the only link between the ground and the two TDRS satellites. The ground station RF equipment provides continuous and simultaneous communications with the two active satellites. Factors such as reliability of the equipment, carrier-to-noise ratio, and rainfall rate could conceivably cause interruptions in these communication links. One of the system requirements is a return link bandwidth of 600 MHz. This is easier to achieve at higher transmission frequencies so that a frequency of 14.6 to 15.2 GHz was chosen. In this range of frequencies, attenuation can vary widely with atmospheric conditions. These effects were considered in arriving at a design for the ground station.

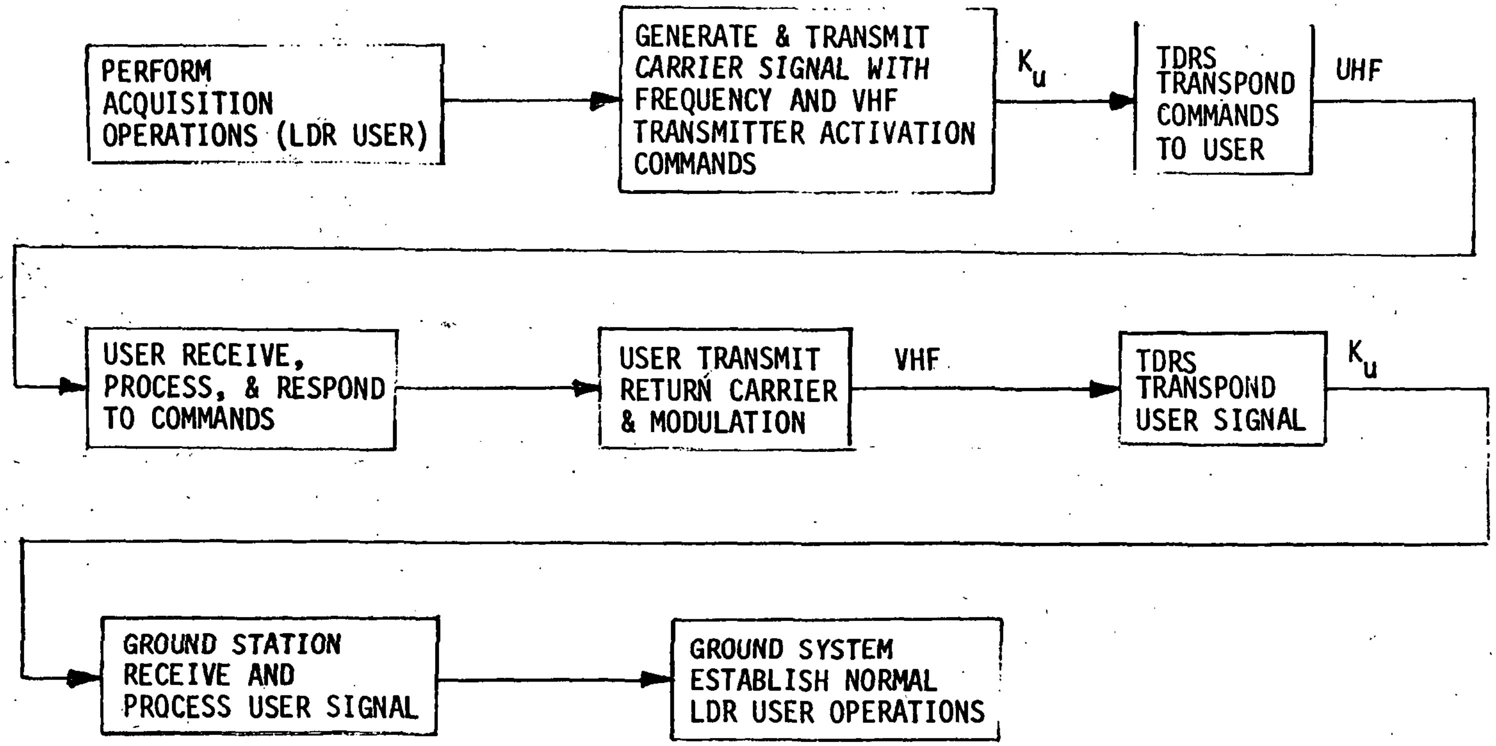
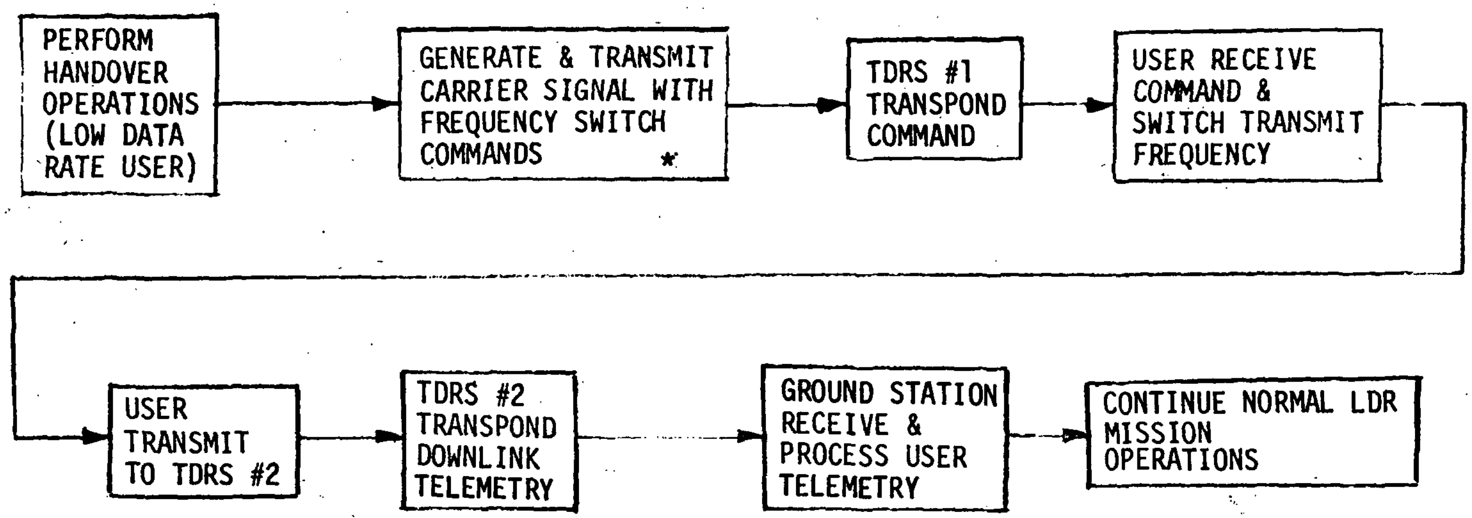


Figure 1-46 LDR USER ACQUISITION OPERATIONS



*TERMINATES WHEN USER REPORTS COMMAND VERIFICATION VIA TDRS #2

Figure 1-47 LDR USER HANDOVER OPERATIONS

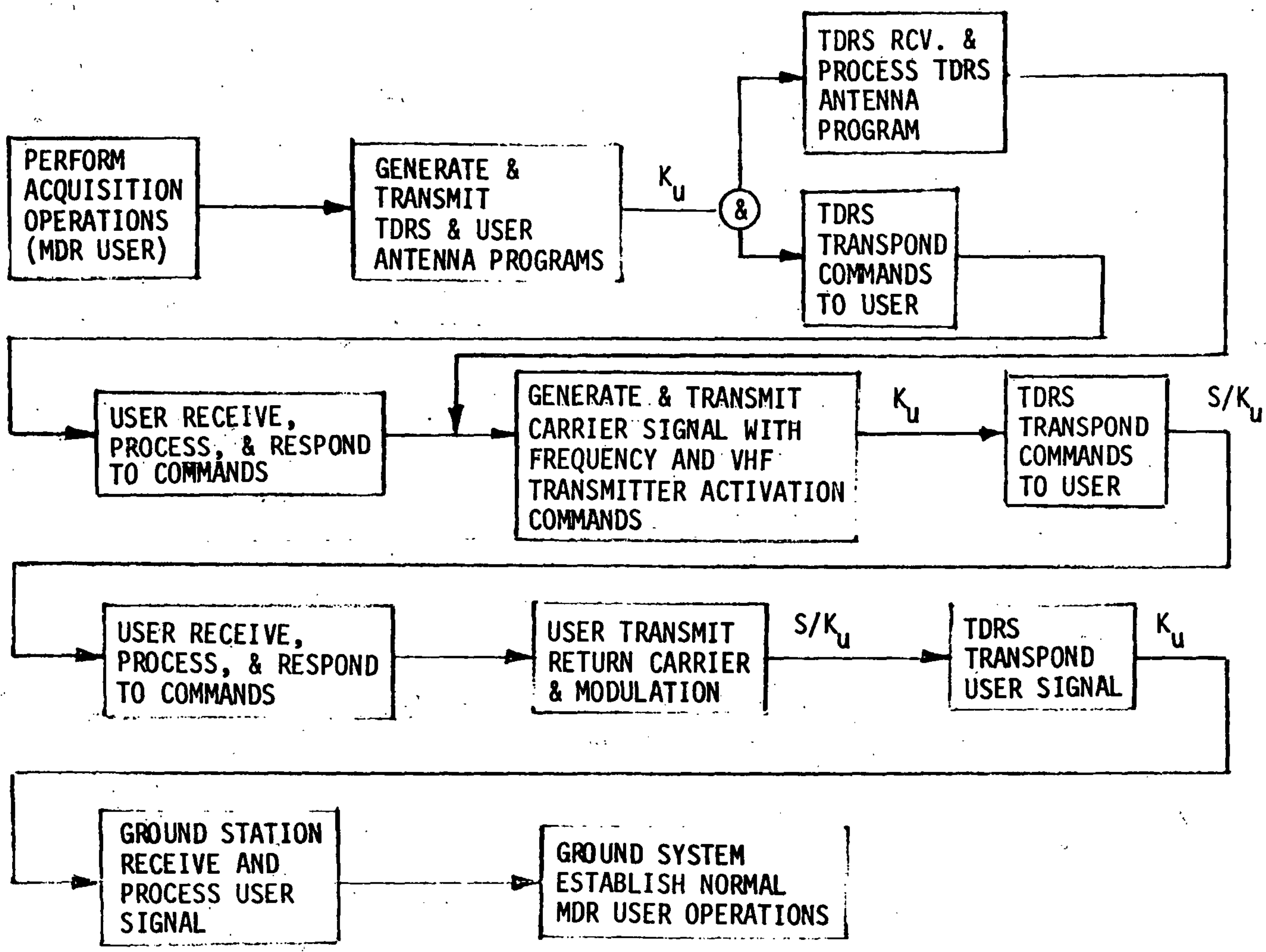
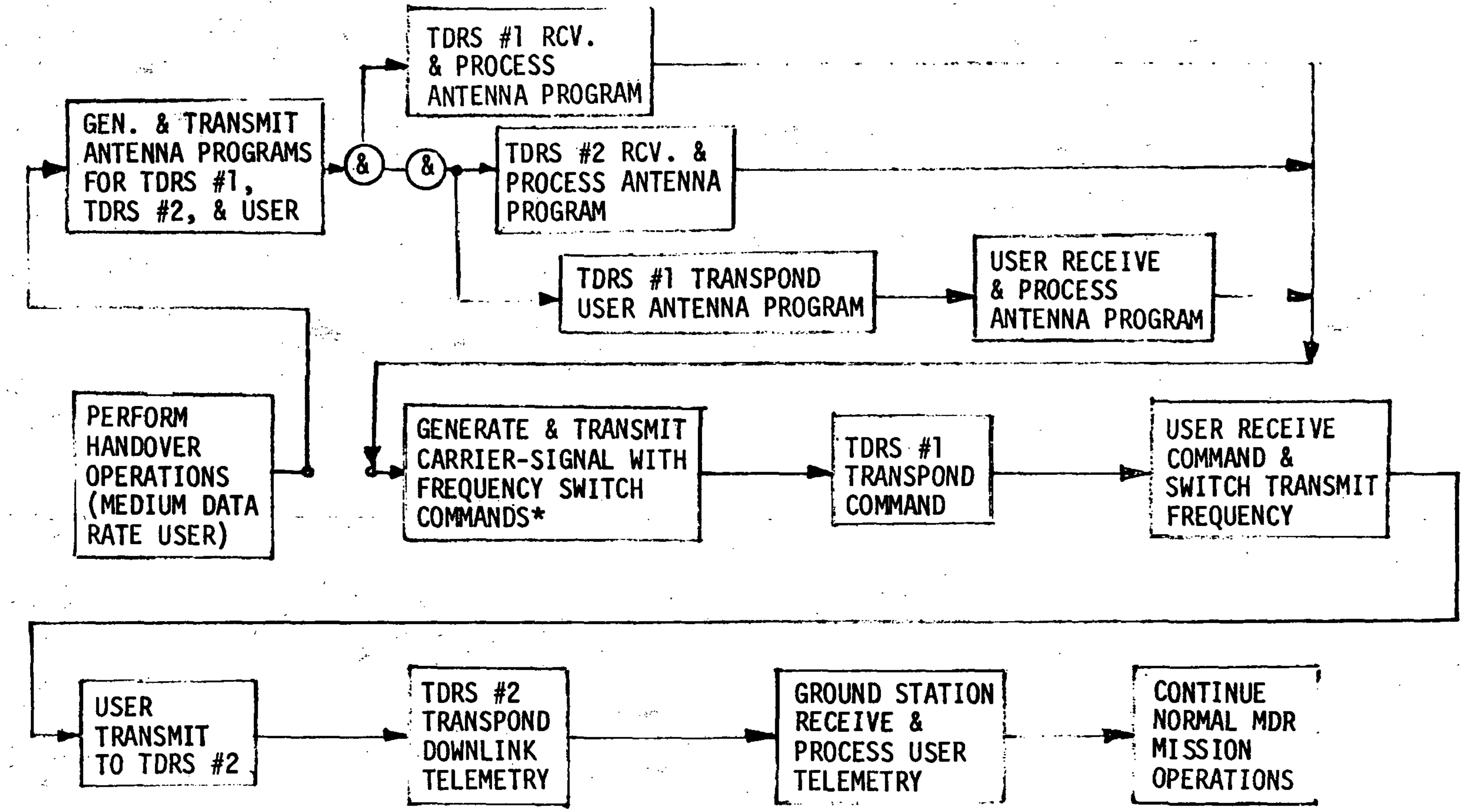


Figure 1-48 MDR USER ACQUISITION OPERATIONS



*TERMINATES WHEN USER REPORTS COMMAND VERIFICATION VIA TDRS #2

Figure 1-49 MDR USER HANDOVER OPERATIONS

The communication link is mostly affected by rain losses, antenna size, receiver sensitivity, and transmitter power. The various effects of these parameters were traded off in selecting a baseline station design. Briefly, the conclusions reached are:

- 17.5 dB rain margin for 99.97 percent reliability
- 60-ft (18.3M) parabolic dish on the ground (one for each TDRS interface)
- Receiver is an uncooled parametric amplifier
- Three each 25 W Klystrons to transmit
- A minicomputer for each AGIPA processor

In addition, since both space-to-earth and earth-to-space links to the TDRS satellites operate at the same frequency, sufficient antenna isolation must be obtained to eliminate cross coupling between the two links. Techniques to provide the required isolation are described in Section 12.

1.13 RECOMMENDATIONS

Several recommendations can be made both with respect to the Part II study, and efforts that should be initiated to complement these studies and support the continued development of the TDRS program. In Part II it is recommended that emphasis be placed on increasing the telecommunications services to include support of the high data rate users both on the standard Delta 2914 launch vehicle and on an upgraded Delta anticipated to be available before the implementation phase. It also is recommended that design of an Atlas-Centaur configuration should not be pursued and greater emphasis be placed on the use of the space shuttle for both multiple launches of configurations similar to those using the Delta and individual launches of higher capacity TDRS configurations.

The design of the TDRS system and the TDRS satellite were carried to a level of detail where certain areas can be identified where further engineering studies or small design/development projects should be conducted.

Continued development of the Adaptive Ground Implemented Phased Array (AGIPA) is essential since this is the optimum and possibly the only viable technique for maintaining communications links at VHF in the face of undefined but possibly high levels of RFI. A laboratory simulation model incorporating the telecommunications subsystem described in this report along with the multimode transponder should be developed to demonstrate overall system performance, and evaluate the effects of various RFI conditions on the system. Preliminary design of the TDRS ground station should be initiated using the conceptual design provided by this study as its basis.

With respect to spacecraft design, it is recommended that the preliminary design of the stabilization and control subsystem be extended to the detail design level and an engineering model be developed. The model would demonstrate the performance characteristics and verify the analytical computer simulation predictions as to spacecraft stability and the effects of slewing large antennas. The design set forth in this study is not only extremely reliable with performance equal to all concepts considered, but has such great operational and design flexibility that it should be considered as being the possible basis for standardizing this subsystem for most unmanned earth orbital spacecraft.

To illustrate the actual simplicity and reliability of the antenna stowage and deployment mechanisms, which cannot be made obvious through the use of engineering drawings and verbal descriptions, it is suggested that a working model be developed.

It is possible that this antenna model could be adapted for use with the above recommendations regarding stabilization and control to realistically simulate the flexible interfaces.

Since it is anticipated that TDR satellites eventually will be launched using the space shuttle, it is recommended that a study be performed to determine the optimum design and plan for evolving from a Delta-launched configuration to a shuttle-launched configuration. Optimum being defined as minimum impact or development required for the transition.